WL-TR-92-2029

PROPULSION/WEAPON SYSTEM INTERACTION MODEL

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July 1992

Final Report for Period December 1983 - September 1987

Approved for Public Release; Distribution is Unlimited

AERO PROPULSION AND POWER DIRECTORATE
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REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY *(Leave blank)*

2. REPORT DATE
July 1992

3. REPORT TYPE AND DATES COVERED
Final, Dec 83 - Sep 87

4. TITLE AND SUBTITLE
Propulsion/Weapon System Interaction Model

5. FUNDING NUMBERS
C-F33615-83-C-2347
PE - 66203F
PR - 3066
TA - 11
WU - 46

6. AUTHOR(S)
Colin Widdison

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
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8. PERFORMING ORGANIZATION REPORT NUMBER
D180-29760-1

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10. SPONSORING/MONITORING AGENCY REPORT NUMBER
WL-TR-92-2029

11. SUPPLEMENTARY NOTES

12a. DISTRIBUTION/AVAILABILITY STATEMENT
Approved for Public Release; Distribution is Unlimited

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*
This document described a computer program used to evaluate advanced airbreathing propulsion payoffs in aircraft performance of future interest to the USAF. The program is required to determine the potential impact of propulsion technology advancements and future weapon system requirements on propulsion concept and cycle selection. A major requirement in such assessments is the evaluation of interaction effects between the engine and airframe. Several scalable airframe "Data bases" were developed to examine a variety of vehicle concepts including a tactical fighter, supersonic interceptor, supersonic cruise missile, logistic transport, lightweight fighter, carrier air vehicle (first stage of a two-stage-to-orbit system) and hypersonic interceptor. A program description including options, mission analysis approach, installation methodology, and program structure is provided. A description of each of the existing data bases including baseline weights, dimensions, and drag characteristics is included. Also, a sample output listing is included in this document.

14. SUBJECT TERMS
Engine/Airframe Computer Performance Model, Propulsion Installation, Mission Analysis

15. NUMBER OF PAGES
152

16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	Unlimited

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LIST OF NOMENCLATURE AND SYMBOLS

A^*	Sonic area, in^2
A	Area, in^2
A_c	Inlet capture area, in^2
A_o	Local stream tube area ahead of the inlet, in^2
A_{OI}	Free stream tube area of air entering the inlet, in^2
C_D	Drag coefficient, dimensionless
C	Sonic velocity, ft/sec
C-D	Convergent-divergent
$C_{DA\text{-}10}$	Afterbody drag coefficient, <u>DRAG</u>, dimensionless
C_{fG}	Thrust coefficient, dimensionless
C_V	Nozzle velocity coefficient, dimensionless
Conv	Convergent
D	Drag
F	Thrust, lb
F_N	Net thrust, lb
F_{NA}	Installed net thrust, lb
F_{gi}	Ideal gross thrust (fully ideal gross thrust) (fully expanded), lb
f/a	Fuel/air ratio, dimensionless
g	Gravitational constant, ft/sec
h	Enthalpy per unit mass, Btu/lb, height in
h_{fan}	Enthalpy of fan discharge flow, Btu/lb
h_{pri}	Enthalpy of primary exhaust flow after heat addition, Btu/lb
h_t	Thrust height, in^2
M	Mach number, dimensionless

LIST OF NOMENCLATURE AND SYMBOLS (Continued)

P	Static pressure, lb/in^2, perimeter, in
P_r	Relative pressure: the ratio of the pressures P_a and P_b corresponding to the temperatures T_a and T_b, respectively, along a given isentrope, dimensionless
P.S.	Power setting
P_T	Total pressure, lb/in^2
Q	Effective heating value of fuel, Btu/lb
q	Dynamic pressure, lb/in^2
R	Gas constant
R,r	Radius, in
R_P	Total pressure recovery
SFC	Specific fuel consumption
SFC_A	Installed specific fuel consumption
T	Temperature
V	Velocity, ft/sec
W	Mass flow, lb/sec
W_{BX}	Bleed air removed from engine, lb/sec
W_C	Corrected airflow, lb/sec
W_f	Weight flow rate of fuel, lb/sec
W	Weight flow rate of air, primary plus secondary, lb/sec
W_G	Primary nozzle airflow rate, lb/sec
T_Z	Temperature correction factor, T_{T2}/T_{STD}
S_{T2}	Pressure correction factor, P_{T2}/P_{STD}
B	Burner efficiency, dimensionless
P	Density, lb/ft^3

SUBSCRIPTS

amb	ambient
AB	afterbody
b	burner
B_x	bleed airflow extracted from the engine
BP	bypass
BLC	boundary layer bleed

1.0 Introduction and Summary

1.1 Introduction

The Turbine Engine Division of the WL Aero Propulsion and Power Directorate frequently carries out studies to determine the potential of propulsion technology advancements and to assess the impact of future weapon system requirements on propulsion concept and cycle selection. To that end, a computer program was written to provide a rapid response capability which gives consideration to diverse mission requirements and accurate propulsion/airframe integration.

To meet these needs the program has the following capabilities:

- estimation of installation effects on engine performance
- ability to calculate airplane performance in any user-defined mission.

The latter capability is dependent upon several items, as follows:

- calculation of airframe weight
- calculation of airframe drag
- calculation of mission performance segments such as CLIMB, CRUISE, etc
- ability to assess the mutual influences between the different technologies involved.

The program was written by adapting a set of existing preliminary design programs into a unified program that can be used to identify airframe/mission interaction effects on advanced propulsion systems.

Criteria for the unified program include:

- rapid turnaround
- interactive capability
- simple to operate
- modular construction.

In conjunction with the program, a data base of several "generic" aircraft configurations was provided. These generic configurations serve as baseline designs that can be used to assess the effects of selecting variations in engine, airframe, and installation parameters. The selection of configurations that have been the subject of serious study assures that the parametric analysis will be carried out realistically.

This document provides an overview of the work accomplished during the PWSIM contract and an introduction to the resulting computer programs. For detailed information, see Figure 1-1.

1.2 Summary

The objective of this research was to develop a computer program for the evaluation of air-breathing propulsion system performance in interaction with aircraft of current or future interest to the USAF. The program was required to allow determination of the potential of propulsion technology advancements and the impact of weapon system requirements on propulsion concept and cycle selection. A major requirement in such assessments is the evaluation of interaction effects between the engines and airframes. The computer program was required to synthesize a variety of vehicle concepts (Figure 1-2).

- a tactical fighter
- supersonic interceptor
- supersonic cruise missile
- logistic transport
- lightweight fighter
- carrier air vehicle (first stage of a two-stage-to-orbit system), and
- hypersonic interceptor.

To meet these objectives the plan of work involved development of two computer programs each consisting of an executive routine, two permanent modules, and an interchangeable "data base" module. Two programs were necessary because of the unique mission requirements for the carrier air vehicle configuration and the design implications imposed on it by the mission requirement of the second-stage vehicle.

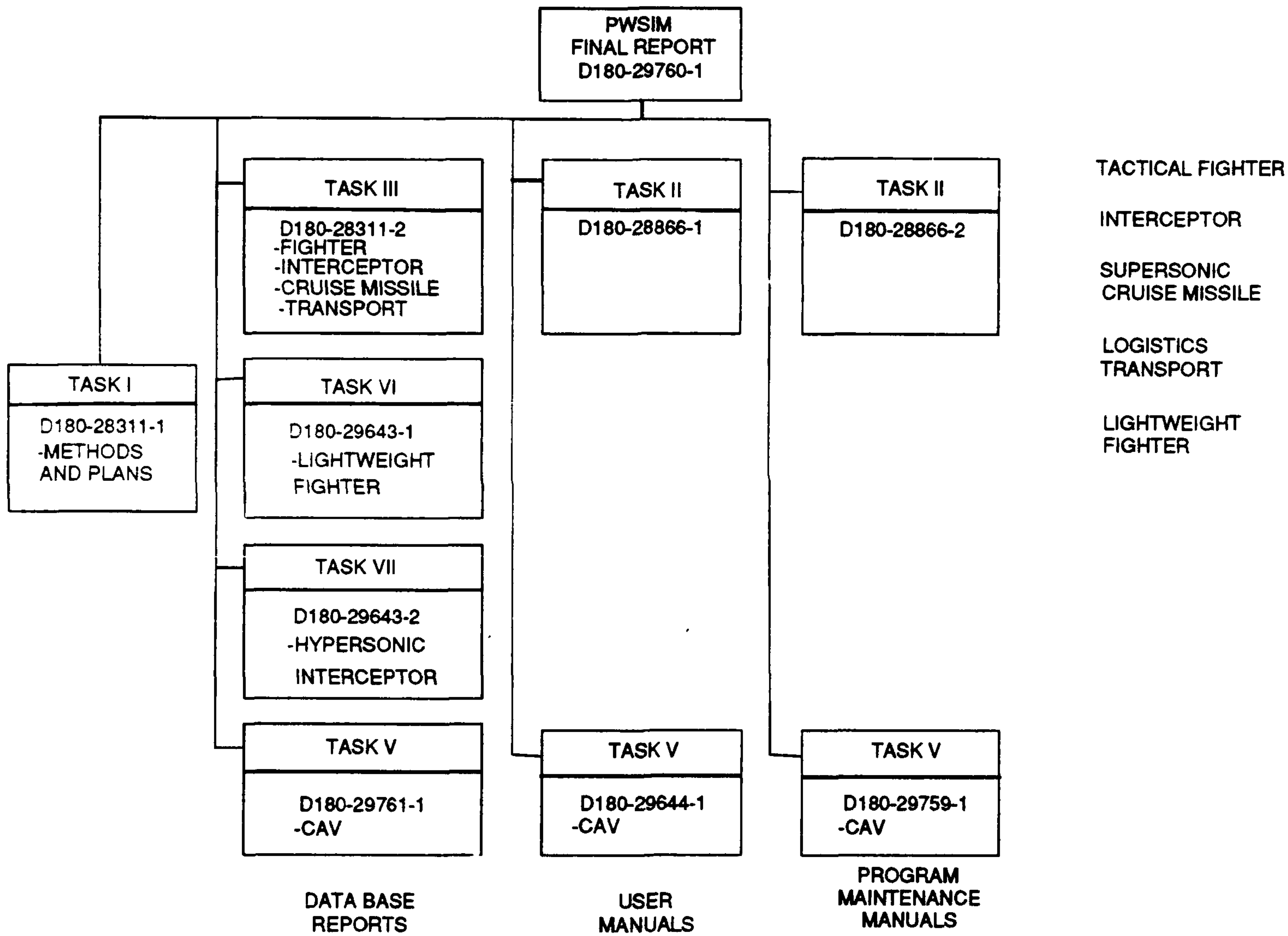
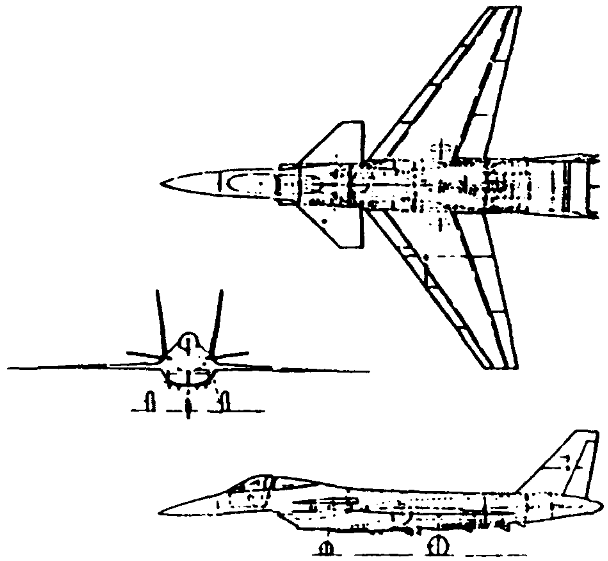
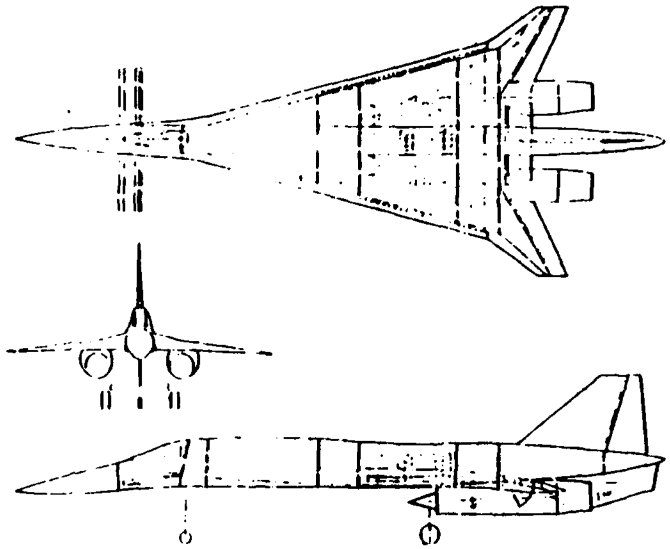


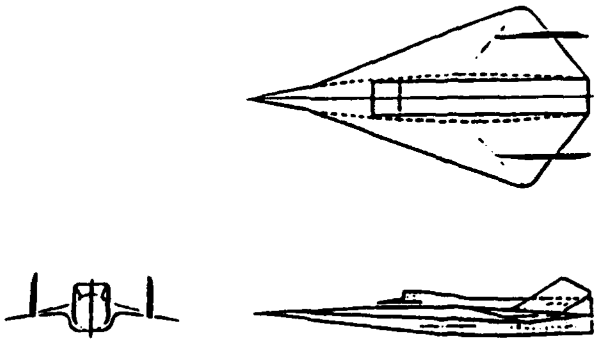
Figure 1-1. PWSIM Documentation



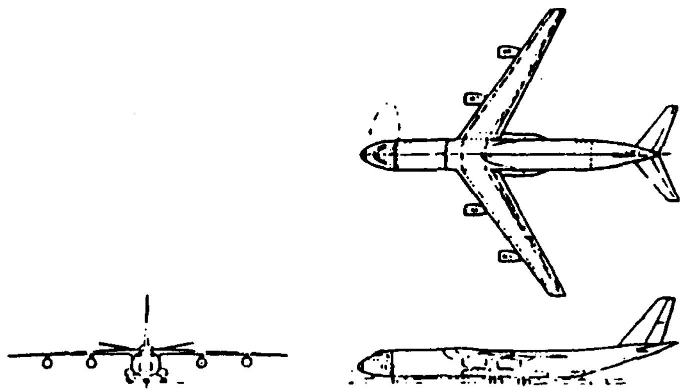
(a) TACTICAL FIGHTER, MODEL 985-420



(b) SUPERSONIC INTERCEPTOR, MODEL 985-430

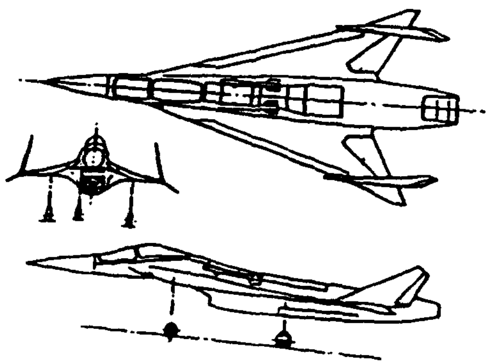


(c) SUPERSONIC CRUISE MISSILE

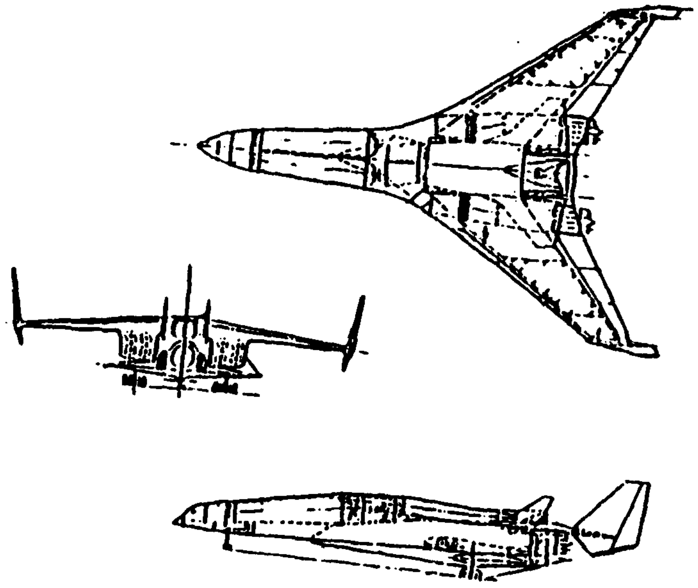


(d) LONG RANGE MILITARY TRANSPORT, MODEL 1046-103

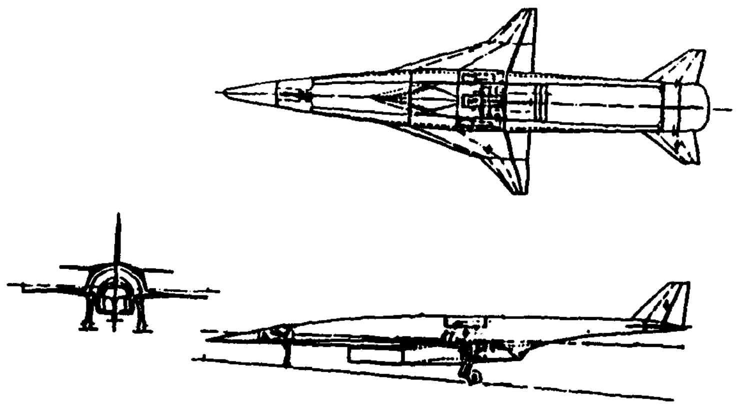
Figure 1-2. Weapon System Configurations



e) LIGHT WEIGHT FIGHTER,
MODEL 985-213



f) CAV/TAV SYSTEM,
MODEL 896-111



g) HYPERSONIC INTERCEPTOR
MODEL 1074-0006

Figure 1-2. Weapon System Configurations (continued)

These programs were mainly derived from several already in use at Boeing, and the bulk of work was associated with adaption and integration of these programs into distinct, compatible modules.

The two permanent modules calculate engine installation effects and airplane size and performance, respectively. To meet the objective of realism (in a preliminary design sense) seven data base modules were developed for use with the program; these represent seven "generic" aircraft configurations. Each data base module contains a data description of a baseline configuration and several routines that allow the program user to scale and modify the baseline with an input data file.

To evaluate the potential of propulsion technology advances, it is necessary to measure their effect on the performance of the system they are likely used in. To be useful, such assessments must be made at the very early stages of technology development to identify promising approaches. Thus, there is need for a rapid response capability which gives consideration to a wide range of configurations, diverse mission requirements, and accurate propulsion/airframe interaction assessment.

The computer program that was developed is capable of calculating either the mission performance of an aircraft of known size or the size of such an aircraft required to perform a specified mission. In executing these calculations, the program takes account of the engine performance (as supplied by the engine manufacturer), engine installation effects of the inlet and nozzle on the engine performance, other engine/airframe interactions, body volume dictated by engine dimensions, wing size influenced by fuel tank volume, etc. and mission requirements. The two main modules of the program deal with engine installation analysis and airplane performance, respectively.

The propulsion installation module is a simplified version of the Boeing Engine Installation Analysis Program (EIAP) and calculates the inlet, nozzle, and aftbody effects on the uninstalled engine performance data supplied by the engine manufacturer.

Inlet effects considered are inlet drag, inlet recovery, and the effects of mismatched inlet air supply and engine air demand. In addition to the effects on engine performance, the inlet routines also allow evaluation of the inlet capture area; this information is supplied to the data base routine to allow engine dimensional considerations to be accounted for in the evaluation of airframe geometry, drag, and weight. Exhaust considerations included are gross thrust and aftbody drag effects.

The airplane performance module (incorporating an airplane sizing option) allows the evaluation of mission performance for an aircraft of known properties (including engines of known installed performance) and also allows "point performance" evaluation at user selected values of weight, altitude, and Mach number.

The mission analysis is performed by calculating aircraft performance during distinct segments (CLIMB, CRUISE, COMBAT, etc.) and linking these segments into a complete mission description.

Seven data base modules were developed for use with the computer program. Each data base represented a typical configuration and was based on an actual preliminary design studied at Boeing.

The data base module consists of:

1. Baseline configuration and modification module that defines the baseline configuration and allows the user to modify (with input data) many of the aircraft design parameters (wing loading, aspect ratio, etc.)

2. A geometry module that evaluates the aircraft dimensions, inlet size, fuel volume, etc.

3. A drag evaluation module that constructs drag tables for use in the performance module, and

4. A weight module that calculates the fuel and operating weights of the aircraft of known gross weight and payload and feeds these numbers to the performance module.

2.0 Program Description

The Turbine Engine Division of the WL Aero Propulsion and Power Directorate is continually engaged in internal studies to determine the potential of propulsion technology advancements and to assess the impact of future weapon system requirements on propulsion concept and cycle selection. This information is required to support Division Long-Range Propulsion planning and program resource allocation in the exploratory and advanced development areas.

To fulfill the need for a rapid response capability which gives consideration to diverse mission requirements and accurate propulsion/airframe integration (and thus provide timely technical assessment program planning information), a comprehensive automated evaluation process is required.

Large, complex computer programs have been developed by the aircraft and propulsion industries to assess future system requirements on advanced weapon system designs. Because of their large computer storage requirements, extensive data base needs and long execution times, these programs could not be efficiently used by the Aero Propulsion and Power Directorate for advanced propulsion assessment work.

The program described in this report combines features of several existing programs for preliminary conceptual analysis into a single program, tailored to specific needs for in-house propulsion assessment. The program is small enough for use interactively but retains sufficient detail in the engineering calculations it performs to assess the effects of engine installation, airframe size and geometry, and mission requirements.

PWSIM is an interactive program for assessing the effects of different engine cycles, engine installations, mission requirements, and airplane geometry on airplane size and weight.

The program is presently able to support seven generic aircraft types, but due to its modular construction, it can accommodate additional configurations.

Configurations currently supported are:

- Tactical Fighter
- Supersonic Interceptor
- Supersonic Cruise Missile
- Long Range Transport
- Lightweight Fighter
- Carrier Air Vehicle
- Hypersonic Interceptor

The program has been coded in extended FORTRAN 77 and runs on the CDC Cyber 175 computer under the NOS 2 operating system with a required field length of about 220K octal words.

The complete program is stored in several different permanent files: one containing the main program executive and the others consisting of libraries of modules which are accessed by the executive. The executive routine accepts the user's input data and controls the sequence of operation to obtain engine performance data and then evaluates airplane size and mission performance. The library modules are of three types:

- propulsion library
- performance library
- data base library.

The propulsion library file contains the routines required to read the uninstalled engine performance data and the inlet and nozzle characteristics and then performs the necessary calculations to evaluate the installed engine performance. The performance library contains the modules needed to calculate the point performance and mission performance of an airplane derived from a data base library.

Several data base libraries are available. Each data base library contains all of the configuration related modules required to define and scale the geometry of a baseline configuration and evaluate its drag polars and operating weight. To accommodate a new configuration, it is necessary to create a new data base library. To execute the program, it is necessary to LOAD the executive program with the performance and propulsion libraries and with an appropriate data base library. Also needed to execute the program are several sets of input data; most of the input data are stored in permanent files, but the user has a certain amount of control via interactive inputs. When the program is executed interactively, prompting messages are provided to the user at the terminal. These messages serve as a guide to allow proper selection of the input required. Most of the interactive inputs are for either selecting calculation options or identifying input data files; some interactive numerical inputs are required when the engine installation option is selected.

2.1 Basic Options

The principal features of the program are:

- An engine installation module that converts engine manufacturer's uninstalled engine performance data into installed performance data by evaluating the internal losses and drag characteristics for the inlet and nozzle/aftbody configuration.

- A set of data files containing inlet and nozzle/aftbody performance maps applicable to suitable engine installation configurations.

- A set of data modules containing data definitions of the generic airplane configurations which allow assessment of the geometric, aerodynamic, and weight characteristics of scaled versions of a baseline aircraft.

- Technology modules that provide rapid and reliable estimates of airplane drag and airframe weight.

- A mission analysis module that allows the user to define almost any practical mission.

- Mission segment modules that use the installed engine performance data and calculated drag polars together with accepted performance methods to assess time, fuel, and distance required to complete the segment.

- The option to scale or "size" an aircraft for a given fixed mission or to find the cruise range or loiter endurance of an aircraft of prescribed size.

- The option to use the engine installation module or use of previously installed engine performance data where appropriate to reduce computation time and cost.

- A choice of interactive or batch operation.

- An output file providing graphics data for a configuration drawing.

This section describes the capabilities and overall operation of the program and outlines the options available to the user.

The program's principal function is to calculate airplane mission performance for an airplane derived from one of several baseline designs.

2.1.1 Mission Performance

Several important parameters affect the performance of an aircraft, namely:

drag
weight
and propulsion system performance.

For an aircraft of known size and geometric proportions, the drag and weight can be estimated (to a satisfactory degree of accuracy) using methods combining analytical and empirical relationships among certain important design parameters.

With such a design the propulsion engineer has an extremely useful tool for assessing the payoff obtained from gains in engine performance, comparing the performance levels of different engines or evaluating the impact of propulsion concept and cycle selection.

The performance level of the propulsion system is not solely a function of the engine or engines selected; it is also strongly dependent upon the way in which the engines are combined with the airframe. For this reason, it is important to assess the effects of the inlet and exhaust systems on the net thrust produced by the engine at any flight condition of interest.

The Propulsion/Weapon System Interaction model computer program (PWSIM) has the capability of evaluating drag, weight (and, therefore, fuel load) and propulsion system performance (including installation effects). It has the further capabilities of calculating airplane performance in terms of the basic components of mission performance such as climb, cruise, loiter, etc. (referred to in this document as mission segments).

In addition to the above, the program provides a facility for calculating the performance of a mission composed of a string of mission segments selected by the user.

Two modes of mission performance are available to the user:

- the aircraft begins the mission at a specified weight (and fuel load) and the program calculates the extent of the mission, the end being determined by attaining a weight equal to the sum of the operating weight, any remaining payload and a specified amount of reserve fuel. (Note: the extent of the mission can have several meanings since the mission requirements may include varying amounts of cruise or loiter.)

- the requirements to achieve a given fixed mission may dictate the use of more fuel than can be carried by an assumed baseline design. The so called "SIZING" option of the program allows scaling of the baseline aircraft to accommodate the extra fuel while taking into account the resulting increases in weight, drag, and engine size.

Figure 2.1-1 shows the interrelationships among the different technologies involved in assessing airplane performance; the message in the box in the lower right corner indicates the two modes of matching the airplane to requirements.

2.1.2 Baseline Designs

The items in Figure 2.1-1 that occupy rectangular boxes are all configuration-dependent. In order for the program to support a wide variety of configurations, each of these areas of the computer program would require sufficient input data (defining the configuration geometry and its influence on drag and weight) to differentiate among the various techniques and methodologies required for the different configurations. Program modules designed to accommodate a wide range of configurations would have the further disadvantage of being complicated because of the large number of decisions required in selecting an appropriate sequence of calculations which results in long execution times and difficulty of maintenance.

In PWSIM these disadvantages have been minimized by the use of "baseline" configurations that have been coded into the configuration-dependent parts of the performance analysis process. By this technique the amount of input data required to define the airframe geometry, weight, and drag is minimized and corresponding program logic is kept relatively simple. A sufficiently large input data set is retained to allow considerable variations from the baseline configuration both in geometry and application.

The apparent disadvantage of being limited to specific configurations is easily overcome because the technology dependent program logic is contained in modules that evaluate airplane geometry, drag and weight, respectively. Thus, an alternate baseline can be "swapped-in" to the program with relative ease.

The program currently supports seven baseline designs:

- Tactical Fighter
- Supersonic Interceptor
- Supersonic Intercontinental Cruise Missile
- Long Range Logistic Transport
- Lightweight Fighter

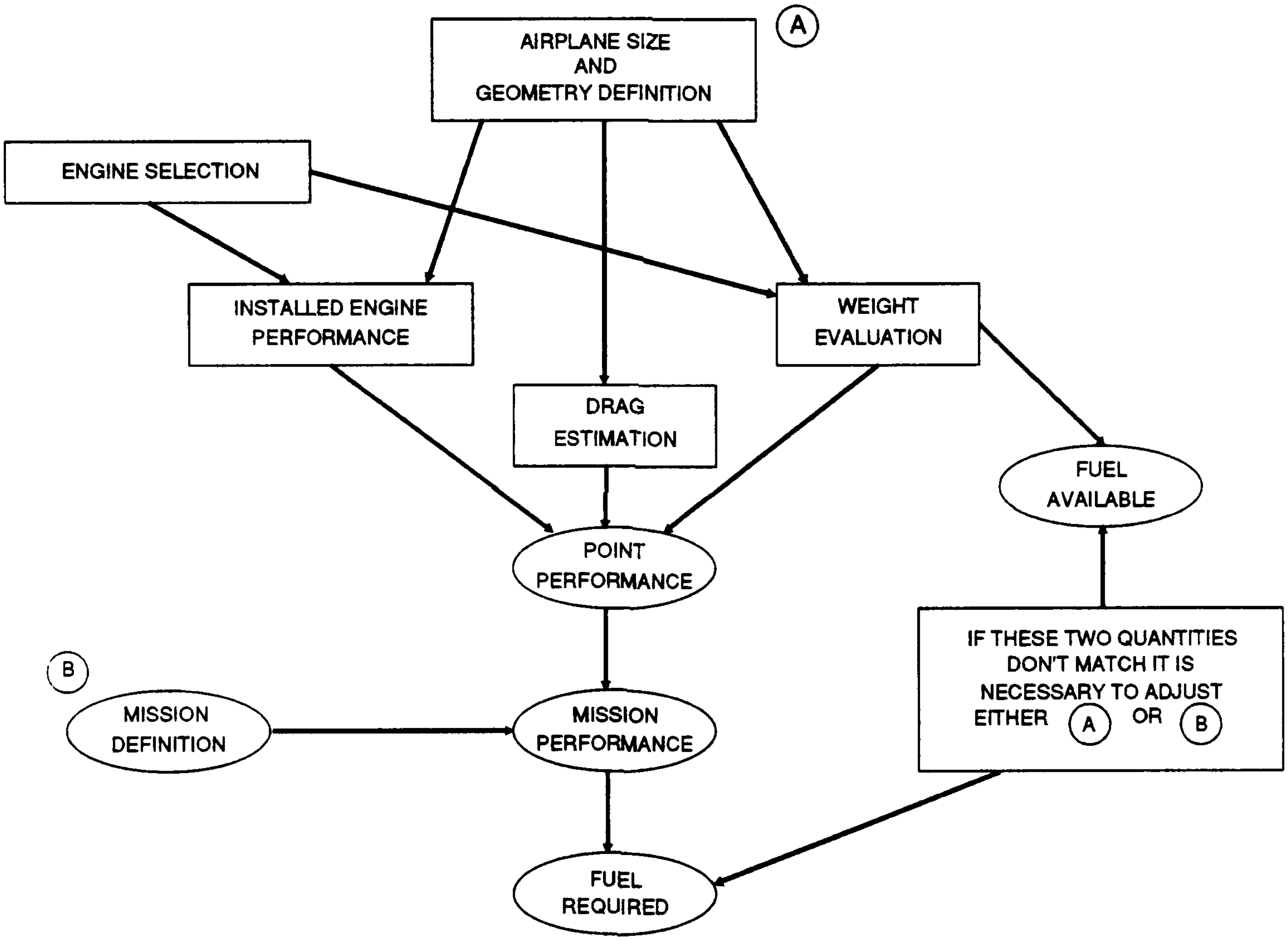


Figure 2.1-1. Airplane Mission Performance Calculation

- Carrier Air Vehicle
- Hypersonic Interceptor

Each was designed by a team of experienced preliminary design engineers to meet specific requirements and as such represents a reliable "point-of-departure" for parametric studies of engine/airframe interactions.

2.2 Mission Analysis

This section describes the method of defining mission profiles.

2.2.1 Background

In many simple performance programs, a mission profile is set up by coding a special subroutine to handle the scheduling of mission segment subprograms, to pass each segment of the input data it needs, and to receive from each segment the output values computed. Complete flexibility as to number, type, and sequence of segments can be achieved in this way; however, the process of setting up or changing a mission program coded in this way is quite cumbersome, since every change requires recompiling and redebugging. In addition, such programs rapidly become expert-dependent, due to the extensive prior knowledge required of the programmer.

Instead of being coded into separate subroutines, mission profiles are defined by a set of input records. At execution time subroutine MISSION schedules segment calculation in the proper sequence, transfers data between segments and handles any iterations required to compute mission distance, time or fuel. All these function are made transparent to the user.

2.2.2 Missions

WHAT A MISSION IS

In the current context, a mission is a flight path that describes the intended usage of the airplane. Missions can be separated into two main classes: fixed performance and variable performance. In a fixed performance mission, all distances and times are fixed, and the result to be computed is the required fuel. In a variable performance mission, the available fuel is fixed and either the mission distance or the mission endurance is to be computed. Variable performance missions are further subdivided into three categories:

- Range Missions - The airplane takes off and lands at different places. The computed range is the distance between the takeoff and landing points.

- Radius Missions - The airplane takes off and lands at the same place. The computed radius is the maximum separation distance of the airplane from the takeoff and landing point.

- Endurance Missions - All distances in the mission are fixed. The result to be computed is the time that can be spent aloft.

PWSIM has the capability of computing all these types of missions.

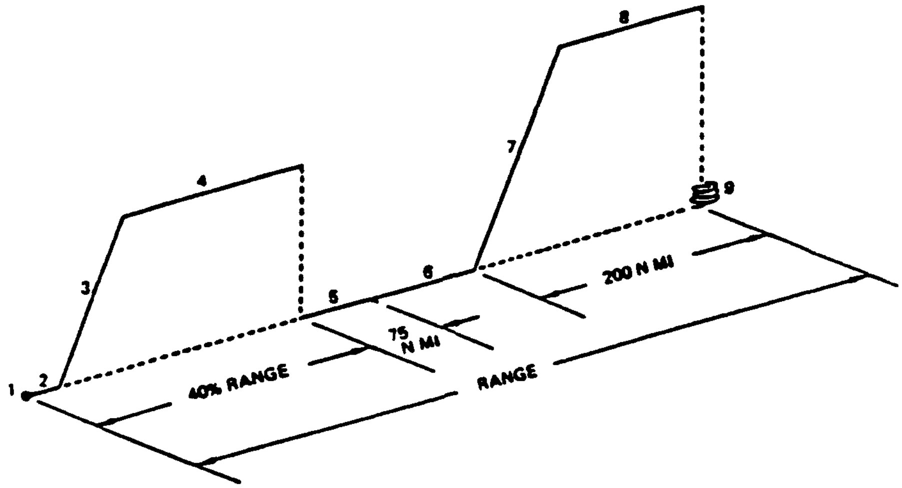
HOW A MISSION IS DEFINED

For analysis, the mission profile is broken down into a number of distinct maneuvers called mission segments. Performance within each segment, that is the distance, time, and fuel required to perform the desired maneuver, is computed from appropriate simplification to the full equations of motion. Individual segments are linked end to end to approximate the desired flight path.

Consider the sample mission profile illustrated in Figure 2.2-1. This profile is typical of high-low-low-high range missions, with cruises of various lengths performed first at altitude then at sea level, and finally at altitude again. Climbs and accelerations are performed for gaining altitude and speed, but no distance credit is taken for descending or decelerating. A total of nine segments are used to describe this flight path.

The data needed to define a mission segment is illustrated in Figure 2.2-1. In general, the definition includes the segment type, the available thrust, the initial and final operating conditions (Mach number and altitude) and for segments such as cruise and loiter, the length of the segment. Other segments, such as acceleration and climb, have lengths determined by the initial and final Mach numbers and altitudes. The performance of the airplane in any one segment is a function of the segment definition and the airplane weight in that segment and hence of the position of that segment in the mission profile sequence.

The mission shown in the example is a variable performance range mission i.e., the amount of fuel in the airplane (at the start of the mission) is fixed and the total mission distance is



1. Takeoff allowance: 5 minutes at intermediate power.

2. Accelerate from Mach 0.3 to Mach 0.5 at sea level.

3. Climb to 25,000 ft. and Mach 0.75.

4. Cruise at 25,000 ft. and Mach 0.75. Elapsed distance at the end of this segment is 40% of the total range.

5. Cruise at sea level and Mach 0.80 for 75 nmi.

6. Cruise at sea level and Mach 0.70. Distance is to be fixed by the total range capability.

7. Climb to 30,000 ft. and Mach 0.70.

8. Cruise at 30,000 ft. Total distance for segments 7 and 8 is 200 nmi.

9. Loiter 20 minutes at sea level and Mach 0.35.

Figure 2.2-1. Sample Mission Profile

to be computed. The distance covered in all segments will be fixed by the segment definition except for segments 4 and 6. These cruises are free to expand and contract until the required mission fuel agrees with the fuel available in the airplane. Distance will be divided up between these two segments so that 40% of the total range is covered before the end of segment 4, and 60% is covered after.

The appearance of a variable length cruise is indicative of a variable performance range or radius mission. A radius mission must have at least two variable distance cruises, of course, since both the outbound and return leg distances are to be computed. For a variable performance endurance mission, all cruise distances must be fixed, and exactly one loiter segment must have a variable time. The extent of the loiter will then be computed so that all available fuel is consumed. For a fixed performance mission, all segments have a fixed length.

2.2.3 Mission Segments

A mission profile is defined to the program by setting up a mission definition file. This file consists of a number of records, each one of which defines a single mission segment. The sequence of the mission segment records in the mission definition file determines the sequence in which the mission segments are executed to form the mission profile. The airplane weight at the start of one segment is set equal to the airplane weight at the end of the preceding segment.

The program has a library of modules to compute fuel used in different types of mission segments including:

- TAXI — operating at a fixed Mach number, altitude, and power setting for a fixed period of time.
- TAKEOFF — accelerating from a standstill to a prescribed percentage over stall speed and climbing to a prescribed height.
- ACCEL — accelerating at constant altitude and power setting from the initial to the final Mach number. Positive or negative acceleration is acceptable.
- CLIMB — climbing from the initial to the final altitude. Available climb schedules include constant equivalent airspeed, constant Mach number, or a combination of the two. Climb schedule may be selected by the module for best rate of climb.

- o CRUISE — performance may be computed for constant altitude cruise or Breguet-type climbing cruise; cruise Mach number and altitude may be selected by the user or computed for best range factor.

- o REFUEL — transfer fuel for tanker to primary mission A/P. Tanker performance simulates the KC-135A.

- o COMBAT — performing a prescribed number of max sustained g-turns. Maneuver load factor may be set by structural limits, maximum lift coefficient or available thrust and may be altered by transfer from the initial to the final Mach number and altitude.

- o DESCENT — dropping from the initial Mach number and altitude to the final Mach number and altitude.

- o LOITER — loiter may be performed at constant altitude or constant lift coefficient. Mach number and altitude may be specified by the user or may be computed for optimum endurance factor.

- o DROP — dropping payload, fuel tanks, or otherwise introducing a weight discontinuity to the mission profile. Drag for the items dropped may be changed by selecting an index that selects one of five arrays of additional drag vs. Mach number.

These are the segments that may be linked together to approximate the mission profile.

HOW A MISSION SEGMENT IS DEFINED

In the most general case, the following data are required to fully define a mission segment.

- o segment type — defines the basic rules governing performance calculation. Examples are: TAXI, TAKEOFF, ACCEL, etc.

- o power setting — refers to a thrust index number defined in the engine deck. For some segment types this index number defines the actual thrust used: TAXI, TAKEOFF, ACCEL, CLIMB, COMBAT and DESCENT. For other segment types, CRUISE, REFUEL and LOITER, this index defines the max available thrust.

- o extent — defines the duration of some segment types. May specify time for LOITER, fuel transferred for REFUEL, time for TAXI, distance for CRUISE, or number of complete turns for COMBAT. For other segment types (TAKEOFF, ACCEL, CLIMB DESCENT) the segment duration is governed by the initial and final Mach numbers and altitude.
- o initial Mach number
- o initial altitude
- o final Mach number
- o final altitude

WHAT THE MISSION SEGMENTS DO

The following paragraphs describe the function of the mission segment performance modules.

In several of the SEGMENTS described below, a flag (TLIMIT) is set to 1.0 if there is insufficient thrust available to achieve the required performance. When program control returns from the SEGMENT calculation to the MISSION subroutine, this value of the flag (TLIMIT) causes printout of the mission history to halt at this segment and print a message to that effect.

<u>TAXI</u>

The TAXI module computes the amount of fuel required to operate at the specified Mach number, altitude and power setting for the time specified (in hours). In this and all subsequent segments, zero is not a valid Mach number.

<u>TAKEOFF</u>

The TAKEOFF segment approximates the time and fuel used in takeoff, that is, between brake release and the end of climbout. Takeoff is approximated by a two-part acceleration to 120% of stall speed, where stall speed is determined by the configuration definition variable CLMAX.

The first part of this acceleration approximates the ground roll up the lift-off speed which is 110% of stall speed. An average acceleration is computed at 0.707 of lift-off speed using thrust determined by the specified power setting and drag computed at lift coefficient CLG, a configuration definition variable. A drag increment for landing gear, Figure 2.2-2, is included.

The second part of the takeoff acceleration, from 110% of stall speed to 120% of stall speed, approximates the climbout. The average acceleration is computed at 115% stall speed, at specified power setting and 1-g lift coefficient. The gear drag increment is included over half this segment.

<u>ACCEL</u>

The ACCEL segment computes the distance, time and fuel consumed in a constant altitude acceleration (or deceleration) between the specified initial and final Mach numbers. Thrust is computed at the input power setting.

Values for altitude and initial Mach number must be supplied to segment ACCEL; however, this segment has two options for computing final Mach number. If "MAX" is specified in place of the final Mach number the segment computes the termination Mach number from placard-limit or thrust-limit conditions. If "MIN" is specified, the final Mach number is computed from the stall margin or thrust limit. Configuration definition variables required to use these options include:

ZMSLM   Max sea level Mach number; defines the constant equivalent airspeed part of the placard.

ZMSUP   Max Mach number at altitude; defines the constant Mach number part of the placard.

CLMAX   Takeoff configuration max lift coefficient.

CLMAXF   Ratio of landing configuration max lift coefficient to takeoff configuration max lift coefficient.

An exceptional condition occurs when insufficient thrust is supplied for acceleration or excess thrust is supplied for deceleration. This exceptional condition is flagged by the ACCEL segment by setting flag TLIMIT=1.

<u>CLIMB</u>

The CLIMB segment computes the distance, time and fuel to climb from the initial Mach number and altitude to the final Mach number and altitude. Thrust is computed at the specified power setting.

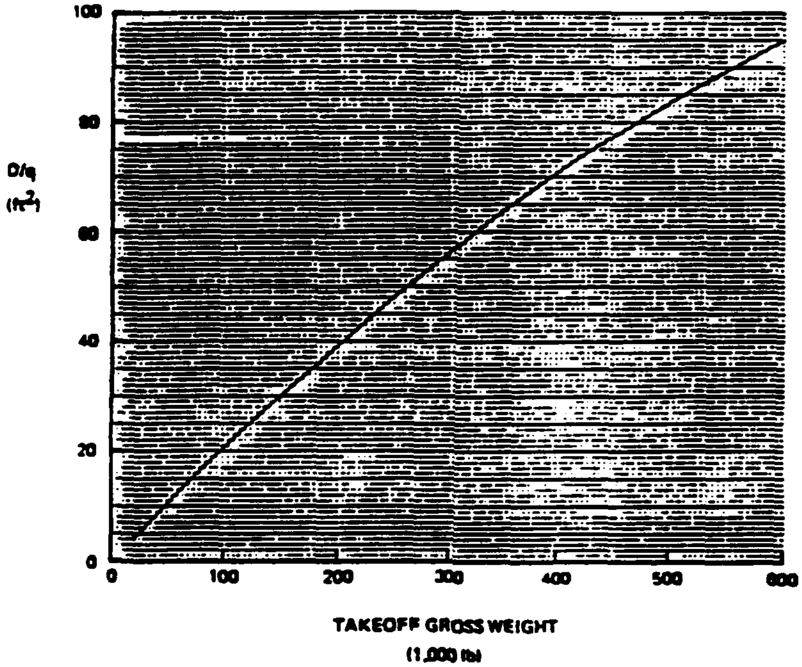


Figure 2.2-2. Estimated Landing Gear Drag Increment

The climb schedule used is determined by the specified Mach numbers and altitudes, as shown in Figure 2.2-3. First, if the equivalent airspeed at the specified final Mach number and altitude is greater than the equivalent airspeed at the specified initial conditions, Figure 2.2-3a, then the climb is performed holding equivalent airspeed constant at the initial value. In this case, the computed final Mach number may not be equal to the specified final Mach number. Second, if the specified final equivalent airspeed is less then the specified initial equivalent airspeed and if the specified final Mach number is greater than the specified initial Mach number, Figure 2.2-3b, then a two-segment climb is performed; a constant equivalent airspeed climb is performed until the Mach number equals the final Mach number and then a constant Mach number climb is performed until the altitude is equal to the final altitude. Finally, if the specified final Mach number is less than the specified initial Mach number, Figure 2.2.3c, then the climb is performed holding Mach number constant at the specified initial value. Here again, the calculated final Mach number may not agree with the specified final Mach number.

The climb schedule may be optimized by specifying "OPT" in place of either the initial or final Mach number (or both). The quantity for which "OPT" was specified will then be computed so as to maximize rate of climb at the specified altitude and power setting. Climb schedule determination will then proceed as above.

An exceptional condition occurs when the airplane becomes thrust-limited along the climb schedule before reaching the final altitude. This condition is flagged by setting the flag TLIMIT=1.

<u>CRUISE</u>

The CRUISE segment computes the time and fuel required to cruise the specified distance. The specified power setting defines the maximum available cruise thrust.

The cruise Mach number may be a specified constant or may be optimized by the segment. If a constant is specified, Mach number is held fixed at this value throughout the cruise. If "OPT" is specified in place of the initial Mach number, both the initial and final Mach numbers are optimized independently, and a slight acceleration or deceleration might result.

Three options are available for determining the altitude profile of the cruise. If the initial altitude is a specified constant and the final altitude is not the same constant, then a

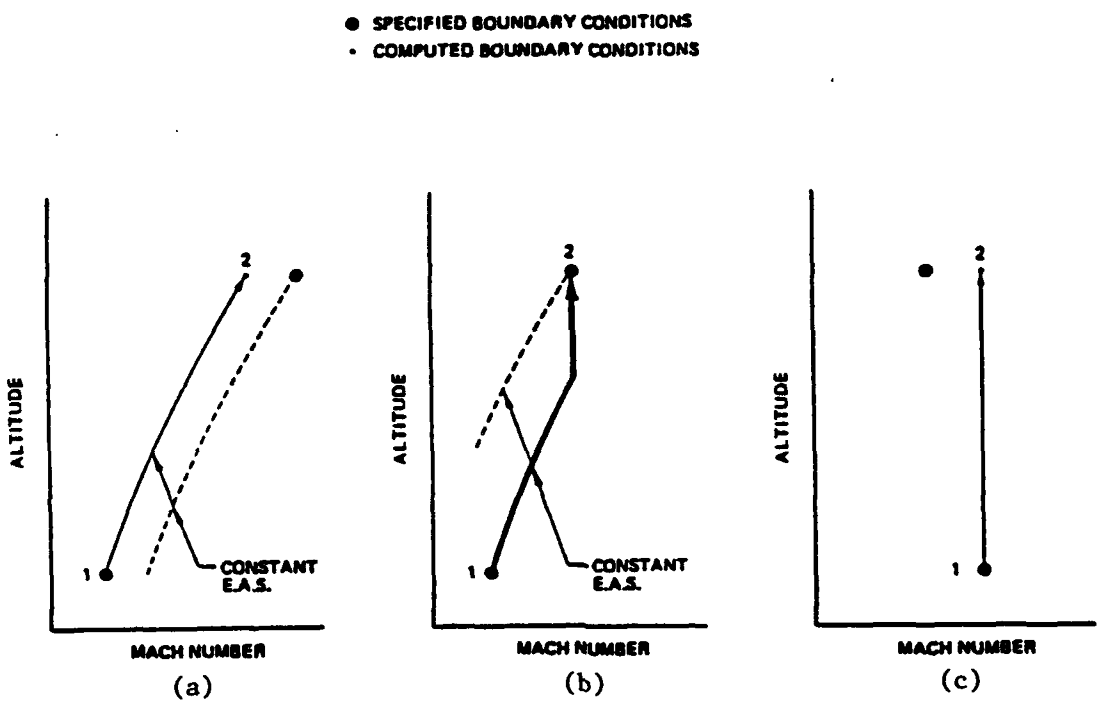


Figure 2.2-3. Climb Schedule Determination

Breguet-type climbing cruise is performed. Next, if the initial and final altitudes are specified equal to each other, than a constant altitude cruise is performed. Finally, if "OPT" is specified in place of the initial altitude, then both initial and final altitudes are independently computed so as to maximize range factor.

If insufficient thrust is available for cruise, the segment value of the flag TLIMIT is set to 1.0.

COMBAT

The COMBAT segment computes the time and fuel required to perform the specified number of 360-degree turns at max sustained load factor. Thrust is computed at the specified power setting; load factor is computed so that the resulting drag agrees with available thrust. The required drag may be increased (or decreased) by the specific energy released (or consumed) in transferring from the initial to the final operating conditions. If final conditions are not specified, they are taken to be the same as the initial conditions.

If insufficient thrust is available at the specified power setting, this is indicated by the flat TLIMIT=1.0.

DESCENT

The DESCENT segment computes the distance, time, and fuel used in descending from the initial Mach number and altitude to the final Mach number and altitude. During the descent, the rate of change of speed with altitude is held constant.

Optionally, the user may specify "MIN" in place of the final Mach number. In this case, the final Mach number will either be 120% of the landing stall Mach number (as determined by configuration definition variables CLMAX and CLMAXF) or the lower thrust limit Mach number, whichever gives the higher value.

REFUEL

The REFUEL segment computes the distance, time, and fuel consumed in receiving the specified weight of fuel from a tanker. The fuel transfer rate and downwash velocity simulate the KC-135A tanker; however, no check is made as to whether the KC-135A could operate at the specified Mach number and altitude or could transfer the required weight of fuel.

The user may specify "MAX" in place of the weight of fuel to be transferred. This signals the segment that fuel is to be transferred until the weight of the airplane is brought up to TOGW, a configuration definition variable.

The final segment weight is restricted to the weight at which the airplane would become thrust limited at the segment Mach number, altitude and power setting. The segment limits the fuel transfer to no more than the amount that would bring the airplane weight up to the thrust limit weight.

LOITER

The LOITER segment computes the fuel required to operate for the specified number of hours at the operating conditions defined. The specified power setting defines the maximum thrust available.

The loiter Mach number may be a specified constant or may be optimized by the segment. If a constant is specified, the entire loiter is performed at this Mach number. If "OPT" is specified in place of the initial Mach number, the initial and final Mach numbers are optimized independently and a slight acceleration or deceleration may result.

Three options are available for determining the altitude profile of the loiter. If the initial altitude is given as a constant and the final altitude is not given as that same constant, then the segment is performed holding W/δ constant at the initial value. Second, if the initial and final altitudes are the same constant, then the entire segment is performed at the constant altitude. Finally, if "OPT" is specified in place of the initial altitude, then both the initial and final altitudes are optimized.

If insufficient thrust is available to perform the loiter, then the segment thrust limit flag is set (TLIMIT=1.).

DROP

The DROP segment provides a way to introduce a weight or drag discontinuity into the mission profile. When the DROP is encountered, weight is decremented by the specified number of pounds and drag is incremented by a value found in one of five arrays of D/q vs. Mach No. selected by the value of INDEXST. No distance, time, or fuel is used by a DROP segment.

2.3 Propulsion Installation Methodology

2.3.1 Introduction

The Engine Installation Analysis Program (EIAP) has been designed to execute on the Boeing Computer Services (BCS) EKS computer system. It is an overlaid program, which is written entirely in FORTRAN IV, and occupies 130K core locations when resident in the computer. The program is interactive in nature in that it asks the user questions in order to input data and to attach files needed for execution.

The program is a suboverlay of the Propulsion/Weapon System Interaction Module, that computes installed engine performance based on a set of engine library maps and an "uninstalled" engine data file. The map library consists of sets of inlet and nozzle performance data. Section II contains an overall description of the installation program and a discussion of the procedures used to calculate inlet performance, nozzle performance, and the installed gross thrust. This manual also contains a macro flow chart of the installation module and detailed description of the program subroutines.

The execution of EIAP is discussed at length in the EIAP User's Manual. The manual describes the program's interactive inputs that are required from the user, as well as the tables of inlet and nozzle performance, uninstalled engine data, and drag reference conditions, which must exist prior to execution. In general, the interactive inputs are used to select the following:

1. file of uninstalled engine data to be processed
2. inlet performance maps from map data base
3. nozzle performance maps from map data base
4. inlet capture area sizing criteria
5. nozzle type (axi or 2-D, convergent or con-di) and limit on nozzle exit area (optional)
6. file of drag reference conditions
7. output options.

2.3.2 Structure and Usage

The engine installation analysis program was designed to speed up the process of calculating installed propulsion system performance data while including realistic effects of inlet and nozzle losses due to drag and internal performance. The program was also designed to satisfy two additional criteria: (1) the accuracy of the data generated by the calculation procedure must be suitable for use in preliminary design studies (when detailed knowledge of all geometric features of the design are not known) and (2) the method must reflect the effects of throttle-sensitive changes in inlet and nozzle/aftbody losses.

EIAP was developed from previous propulsion system installation programs. EIAP utilizes a computer-stored library of inlet and nozzle performance characteristics and uninstalled engine data as input to interactively calculate installed propulsion system performance. A chart showing how this computer program is used in a typical preliminary design analysis process is presented in Figure 2.3-1. The calculation of installed propulsion system performance is almost instantaneous if the tabulated performance characteristics of the desired inlet and nozzle/aftbody configurations are available are previously-stored computer files. To provide a readily-available source of inlet and nozzle/aftbody data, a library of inlet and nozzle/aftbody performance characteristics was created that covered a wide variety of possible configurations. During execution of EIAP, these files are attached externally to the program. The user then enters the interactive input commands. The output from the program can be displayed on a terminal or stored on a output disk file for disposition to an off-line printer.

The single most important factor that made it possible to reduce the time required to perform installed propulsion system performance calculations was the extensive use of computerized files. These files contain tables of data representing the nondimensionalized performance characteristics of inlets and nozzles. They allow instant retrieval of inlet and nozzle/aftbody data that can be matched with the uninstalled engine performance data (also contained in a computer file) during the execution of the program. The formats of the inlet and nozzle/aftbody computerized files and the uninstalled engine data were selected to provide a standardized frame work in which either experimental data or the results of analytical calculations could be used. The input format for the data remains constant, but the data that go into the tables can come from various sources depending on the amount of time available for preparing the data and/or the amount of experimental data available. Because data in the input tables can be changed as better data become available, it is possible to improve the accuracy of the installed propulsion system performance calculations as the aircraft development cycle progresses from preliminary design through full-scale flight test.

The installation module consists of calculations that fall into one of two main categories. The first, the inlet procedure, handles the functions of sizing the inlet, matching the inlet input data with engine airflow demand, and obtaining the matched inlet performance parameters from the inlet data tables. Engine

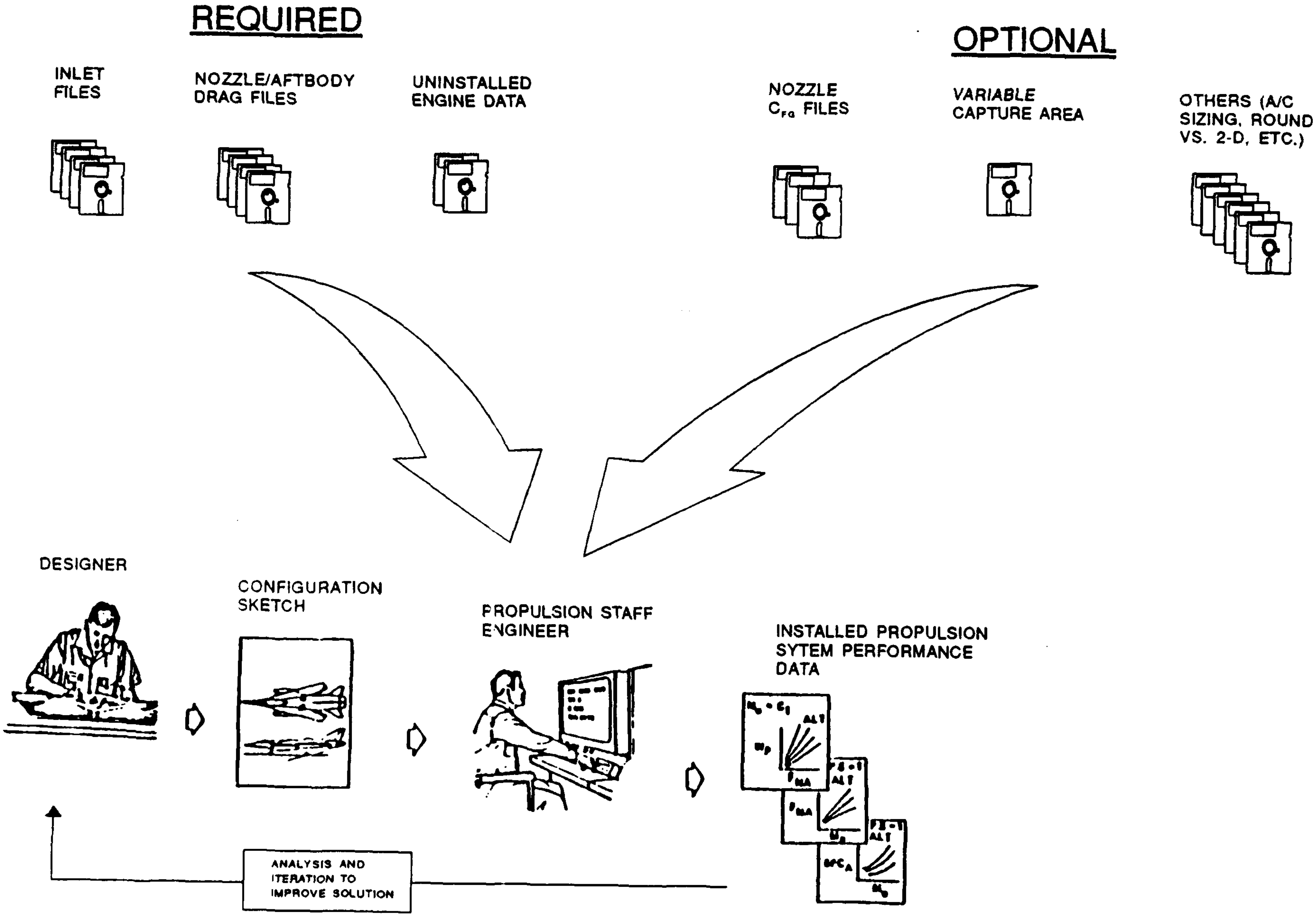


Figure 2.3-1. Preliminary Analysis Process Developed For EIAP

corrected airflow is the matching parameter between engine data and inlet data. The second category, the nozzle procedure, handles the calculation of the nozzle/aftbody drag and nozzle internal performance. Nozzle pressure ratio, P_{T8}/P_o, is used as the matching parameter.

INLET PROCEDURES

The inlet performance procedure of EIAP is considerably longer than the nozzle performance procedure. This is because the individual inlet component drags that contribute to the total inlet drag must be calculated separately. Each of these drags (spillage, bleed, and bypass) must be determined individually as a function of mass flow ratio, which adds to the complexity of the computer program.

INLET PERFORMANCE

The inlet performance maps are input to the program prior to the call to the inlet procedure. This procedure sizes the inlet capture area (if it is required) and converts the inlet performance maps into total pressure recovery and inlet drags that are matched to the corrected airflow demands of the engine.

The operation of the inlet procedure is shown schematically in Figure 2.3-2. The connecting link between the engine data and the inlet procedure is engine operation at a desired inlet mass flow ratio and recovery using the design engine airflow demand. A specified capture area size can be input, if desired, instead of requiring the program to calculate the size.

The inlet input requires three tables of input data which describe the performance characteristics of the inlet. Engineering data obtained from wind tunnel tests and theoretical calculations are used to obtain the inlet performance characteristics. The format of the inlet tables is shown in Figure 2.3-3. The nomenclature for the tables is shown in Figure 2.3-4. Together, the tables form a map, which is entered into the EIAP map library.

The inlet procedure recognizes two modes of inlet operation: low speed mode and high speed mode. The low-speed mode is used only at very low Mach numbers, e.g., takeoff conditions, when only high engine power settings are likely to be of interest and inlet drag is negligible. The high speed mode is used over the remaining Mach number regime. The EIAP calculations of recovery and drag are illustrated in Figure 2.3-5. The required performance maps are input as tables, as indicated. In this mode and the low-speed mode, recovery is read directly out of Table

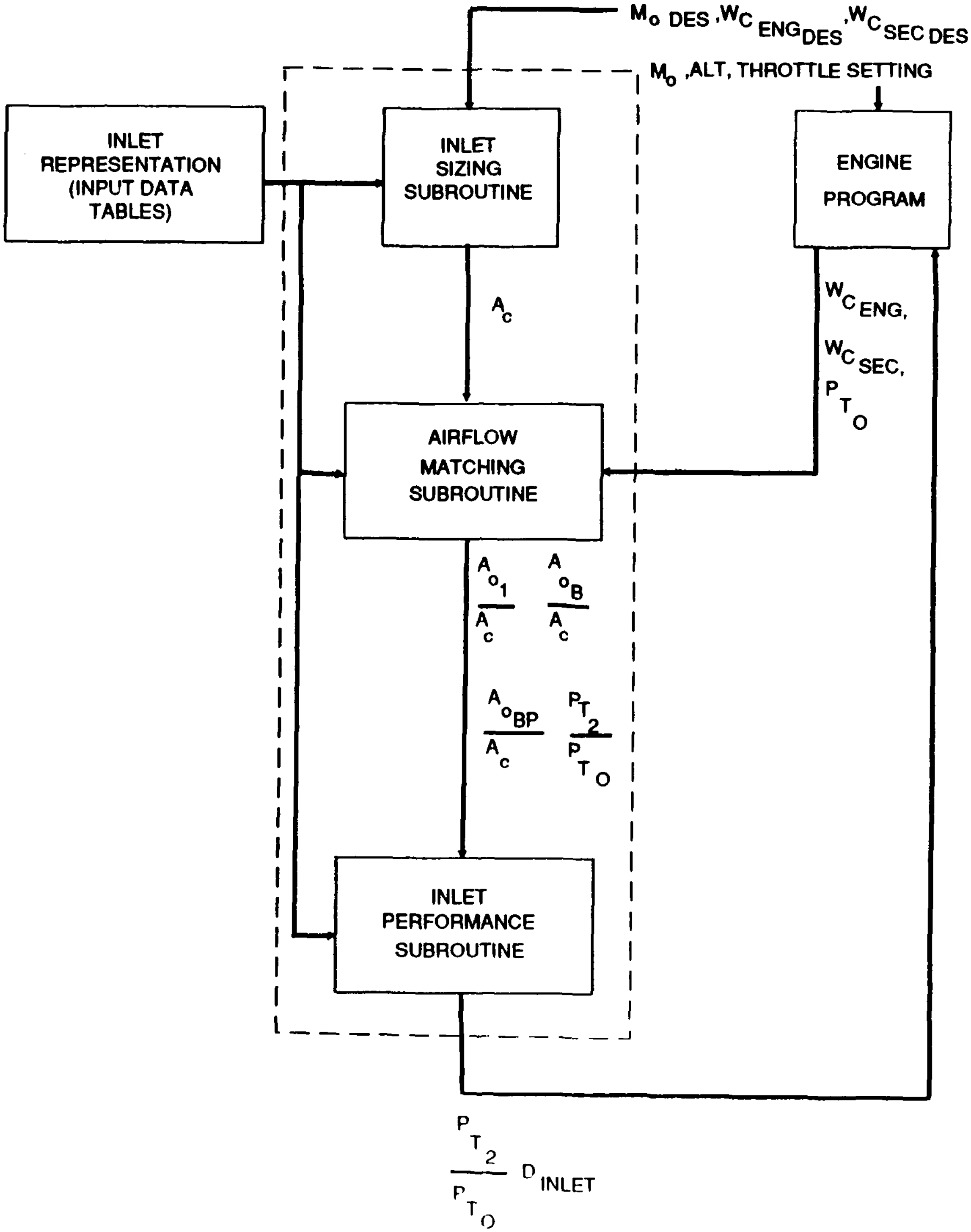


Figure 2.3-2. Inlet Procedure

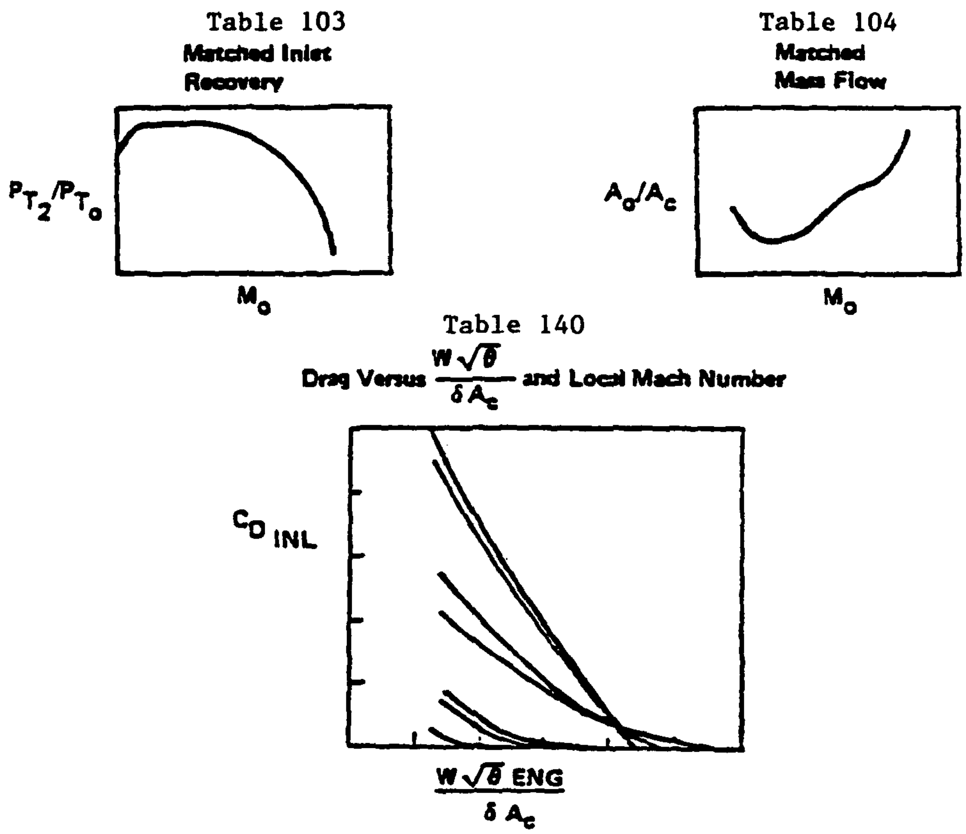


Figure 2.3-3. New Inlet Performance Tables

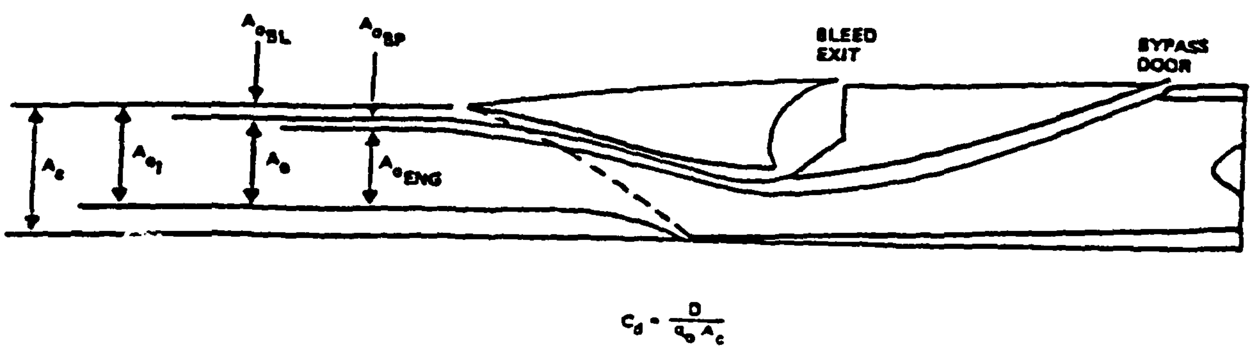


Figure 2.3-4. Inlet Nomenclature

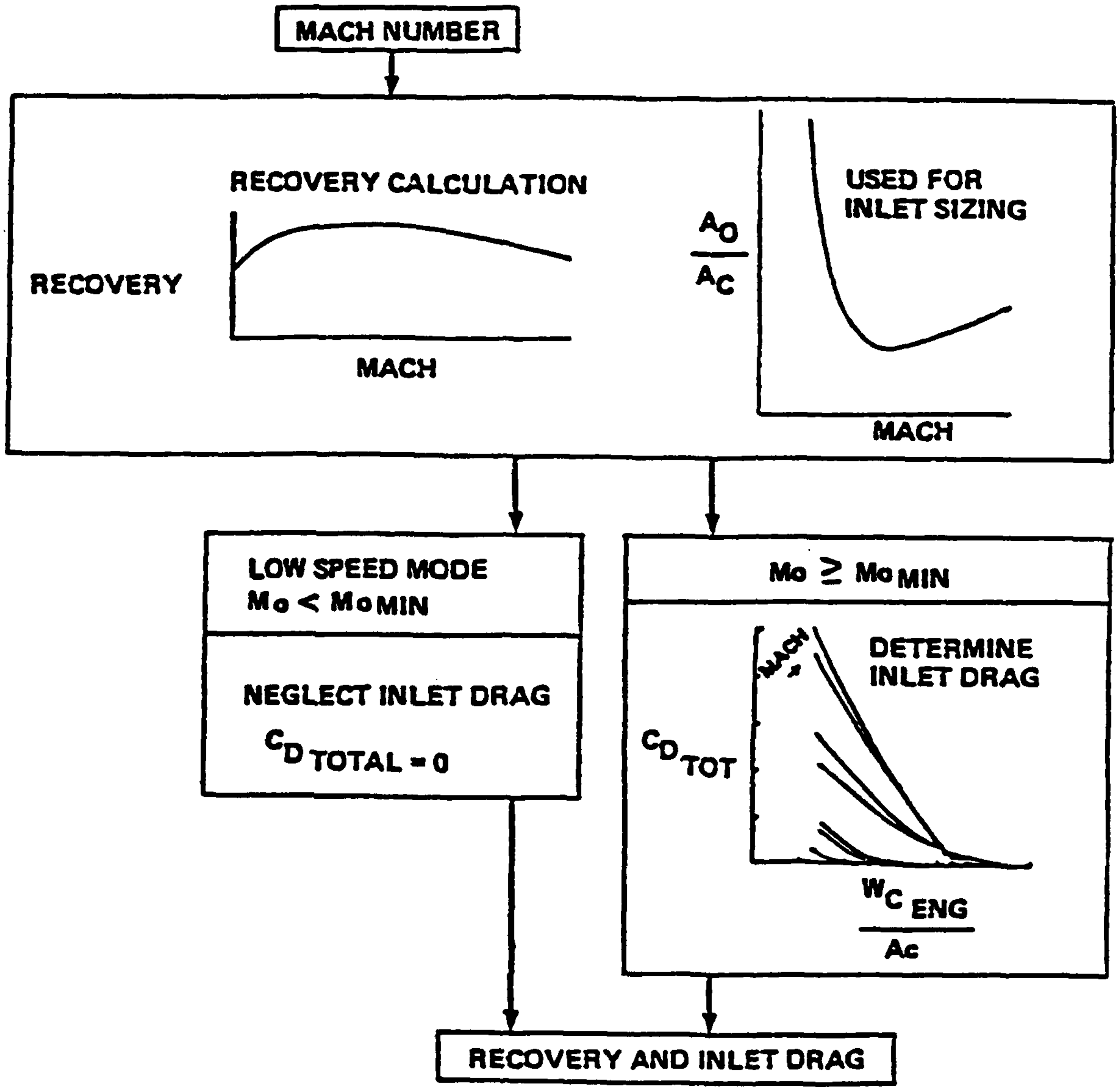


Figure 2.3-5. Proposed Inlet Performance Calculation

103 as a function of local Mach number only. The inlet drag, including spillage, bleed, and bypass drag, is found in Table 140 as a function of Mach number and the ratio of inlet corrected airflow and inlet capture area (WCAC). WCAC is calculated from the engine demand, inlet recovery, and the inlet supply mass flow ratio (found in Table 104 as a function of Mach number). The minimum Mach number entered in Table 140 is used as the minimum value for which the high speed mode is used.

If the corrected airflow delivered by the inlet is inadequate to meet the engine demand at the scheduled recovery, the program will permit the inlet to operate at an excessive supercritical margin. The recovery will be lowered sufficiently to match the engine corrected airflow demand, and an appropriate message will warn the user of an undersized inlet.

INLET SIZING

The inlet sizing procedure in the computer program determines the inlet capture area required to match the largest engine airflow demand at each Mach number. From these calculated inlet sizes, the largest required size is selected as the inlet capture area. For sizing calculations, an input curve (Table 104) of recommended (matched) inlet airflow variations (A_o/A_c) vs. M_o and an input curve (Table 103) (see Figure 2.3-3 for these tables) of recommended (matched) inlet total pressure recovery vs. M_o are used to determine the required capture area variation with Mach number. These parameters are used in the following equation to calculate area, A_C:

$$A_c, in^2 = \frac{A_{oENG}}{(A_o/A_c)\,MATCHED} = \frac{W\sqrt{\theta}\,\frac{P_{T_2}}{P_{T_{O_{MATCHED}}}}\,\frac{A}{A*_o}}{0.343\,(A_o/A_c)_{MATCHED}}$$

INLET RECOVERY CORRECTION

The engine input provides the required data for inlet drag, inlet recovery, nozzle/aftbody drag, and nozzle coefficient calculations. The engine section of EIAP calculates only the changes in internal performance due to changes in inlet recovery. Changes in inlet recovery produce a directly proportional change in nozzle pressure ratio, airflow, and fuel flow because the nozzle throat area does not change. Furthermore, it is assumed that engine data are calculated with MIL-STD-5008B recovery. All inlet recovery changes are made relative to that value unless the user inputs a different reference recovery. Thermodynamic data from Keenan and Kaye tables have been "curve-fitted", and subroutines are provided to calculate the thermodynamic properties of the exhaust gases.

The gross thrust calculation procedure is as follows: for each altitude, Mach number, and power setting, the net thrust (F_N), fuel flow (W_F), corrected airflow ($W\sqrt{\theta_2}/\delta_2$), nozzle throat area ($A_8$), nozzle exit area (A9), and nozzle thrust coefficient (CFG) are given.

A standard atmosphere and MIL Standard 5008B inlet recovery are used to calculate the airflow at the engine face. Gross thrust is found for the given engine data (before any changes in inlet recovery) by the following equations.

$$F_{G_{OLD}} = F_N + \frac{W_2 V}{g}$$

The desired inlet recovery is obtained from the inlet procedure, and the engine gross thrust is first calculated with MIL Standard recovery and then with the calculated recovery. To calculate engine gross thrust, the engine corrected airflow remains constant for any change in inlet recovery, and at any given power setting, the nozzle exhaust areas and burner fuel-air ratio also remain constant. The engine performance for any change in inlet recovery is calculated by the following relations:

$$(W_8)_{RF} = W_8 \frac{(P_{T_2}/P_{T_o})}{(P_{T_2}/P_{T_o})_{MIL\ 5008B}}$$

$$(W_F)_{RF} = W_F \frac{(P_{T_2}/P_{T_o})}{(P_{T_2}/P_{T_o})_{MIL\ 5008B}}$$

$$(W_2)_{RF} = W_2 \frac{(P_{T_2}/P_{T_o})}{(P_{T_2}/P_{T_o})_{MIL\ 5008B}}$$

$$(P_{T_8}/P_O)_{RF} = P_{T_8}/P_O \frac{(P_{T_2}/P_{T_o})}{(P_{T_2}/P_{T_o})_{MIL\ 5008B}}$$

(RF - Recovery factor)

After the above quantities are computed, the corrected quantities $(W_8)_{RF}$, $(W_F)_{RF}$, $(W_2)_{RF}$ and $(P_{T8}/P_o)_{RF}$ are used to compute a new gross thrust, F_{G2}. This new gross thrust and the gross thrust, F_{G1}, calculated using the same subroutines and the uncorrected (MIL 5008B) quantities (W_8, W_F, W_2, P_{T8}/P_o) are used to compute a ratio, F_{G2}/F_{G1}. This ratio is then used to obtain the new value of gross thrust, F_{GNEW} which is found by the ratio:

$$F_{G_{NEW}} = F_{G_{OLD}} \frac{F_{G_2}}{F_{G_1}}$$

The ratio procedure is used to minimize any inaccuracies that may be caused by assuming burner efficiency (η_B) is constant for all engine operating conditions.

The net thrust and fuel flow after correction for inlet recovery are:

$$F_{N_R} = F_{G_{NEW}} - \frac{WV}{g} \frac{RF}{RF_{MIL}}$$

$$W_{F_R} = W_F \frac{R_F}{R_{F_{MIL}}}$$

and the installed propulsion system thrust and SFC are:

$$F_{N_A} = F_{N_R} - D_{INLET} - D_{NOZ} + D_{NOZ\ REF}$$

$$SFC_A = \frac{W_{F_R}}{F_{N_A}}$$

NOZZLE PROCEDURE

The purpose of the nozzle/afterbody drag and CFG input data and calculation procedures is to calculate nozzle internal losses and nozzle/afterbody drag.

NOZZLE/AFTERBODY DRAG

The nozzle/afterbody drag is computed using tables which represent the afterbody drag characteristics (Figure 2.3-6) as a function of P_{s9}/P_o, A9/A10, M_o, external input geometry and engine data. Parameters obtained from the engine calculations include nozzle throat area, nozzle pressure ratio, freestream conditions, and ideal gross thrust. An essential geometry input is the nozzle exit area, A_9, which is required for boattail drag computation. This parameter is obtained in one of two ways:

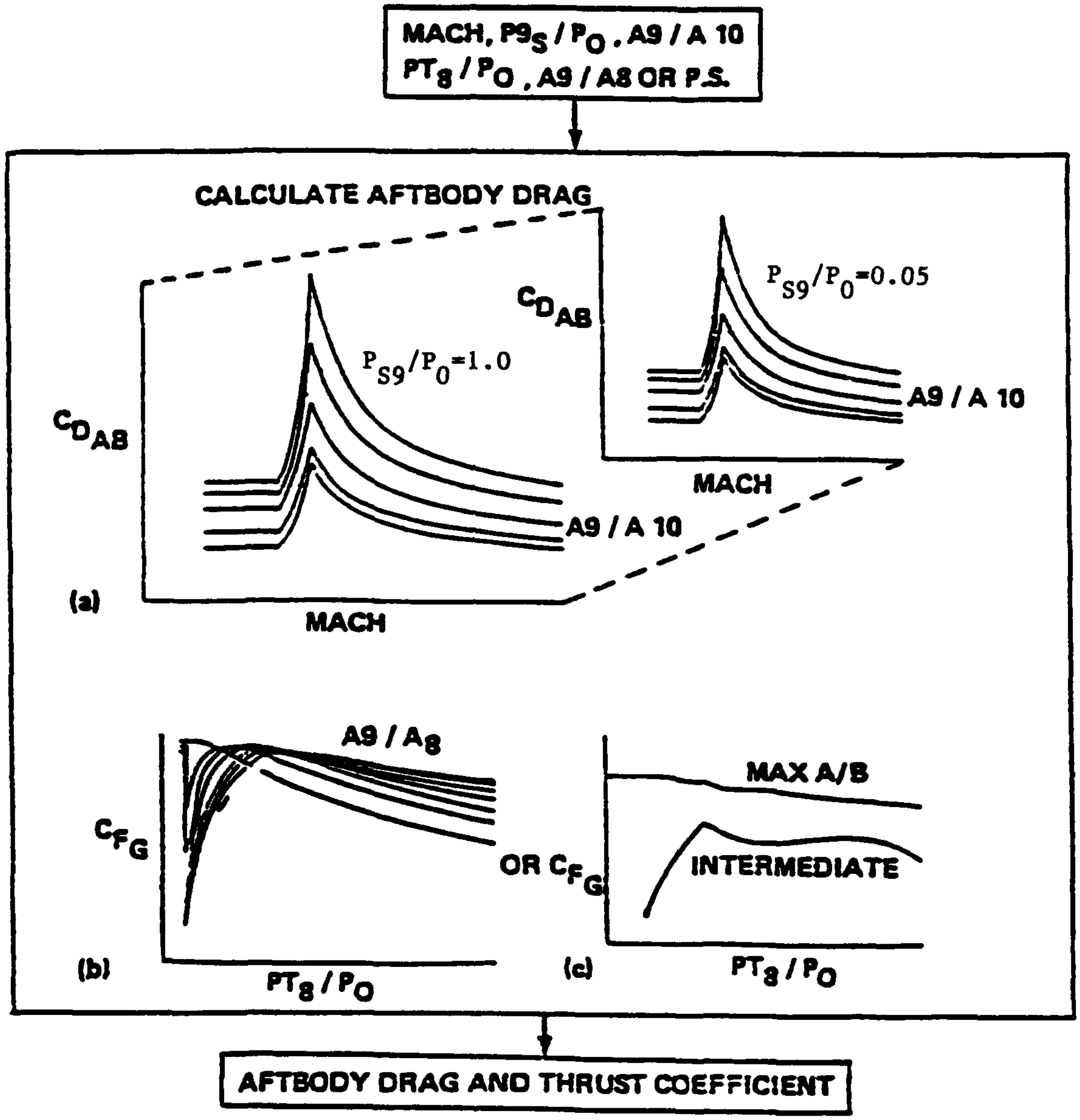


Figure 2.3-6. Nozzle Performance Calculation

1. From the engine input data when the existing axisymmetric nozzle data are used,

2. From a calculation of fully expanded A_9 as a function of nozzle total pressure ratio.

The nozzle/aftbody drag coefficient is shown in Figure 2.3-6(a). The drag coefficient is obtained as a function of the ratio of nozzle exit area to maximum cross-sectional area, A_9/A_{10}, and free-stream Mach number, and nozzle exit static pressure ratio, P_{s9}/P_0. An illustration showing the nozzle aftbody drag procedure is presented in Figure 2.3-7.

NOZZLE GROSS THRUST COEFFICIENT

The nozzle gross thrust coefficient (CFG) tables are used to provide a means for correcting uninstalled engine data for the effects of nozzle internal performance that is different from the nozzle internal performance used in generating the uninstalled engine data. The use of a thrust coefficient table is optional. If no table is used, however, the program will calculate an adjustment to the CFG of the uninstalled data and use this new CFG to find the new installed thrust. The adjustment is only made if the nozzle conditions result in over or under expansion losses.

Two different types of data input formats are provided for the CFG tables. They are shown in Figures 2.3-6(b) and (c). The first table shows nozzle gross thrust coefficient as a function of nozzle static pressure ratio and area ratio. A_9/A_8 is calculated from tabulated input values provided along the second table; however, the nozzle gross thrust coefficient is input as a function of nozzle total pressure ratio and maximum afterburning and intermediate (dry) power settings. This input data format is based on the use of a variable area nozzle which is scheduled to provide an optimum variation of area ratio as a function of nozzle pressure ratio. The engine power setting and nozzle pressure ratio are obtained from the engine input data in the engine performance calculations.

NOZZLE REFERENCE CONDITIONS

The calculated installed propulsion system performance data include the throttle-dependent inlet and nozzle/aftbody losses. To determine the throttle-dependent portion of the nozzle/aftbody drag to be included as a loss to the propulsion system performance, a reference condition has been established for the nozzle/aftbody drag as follows:

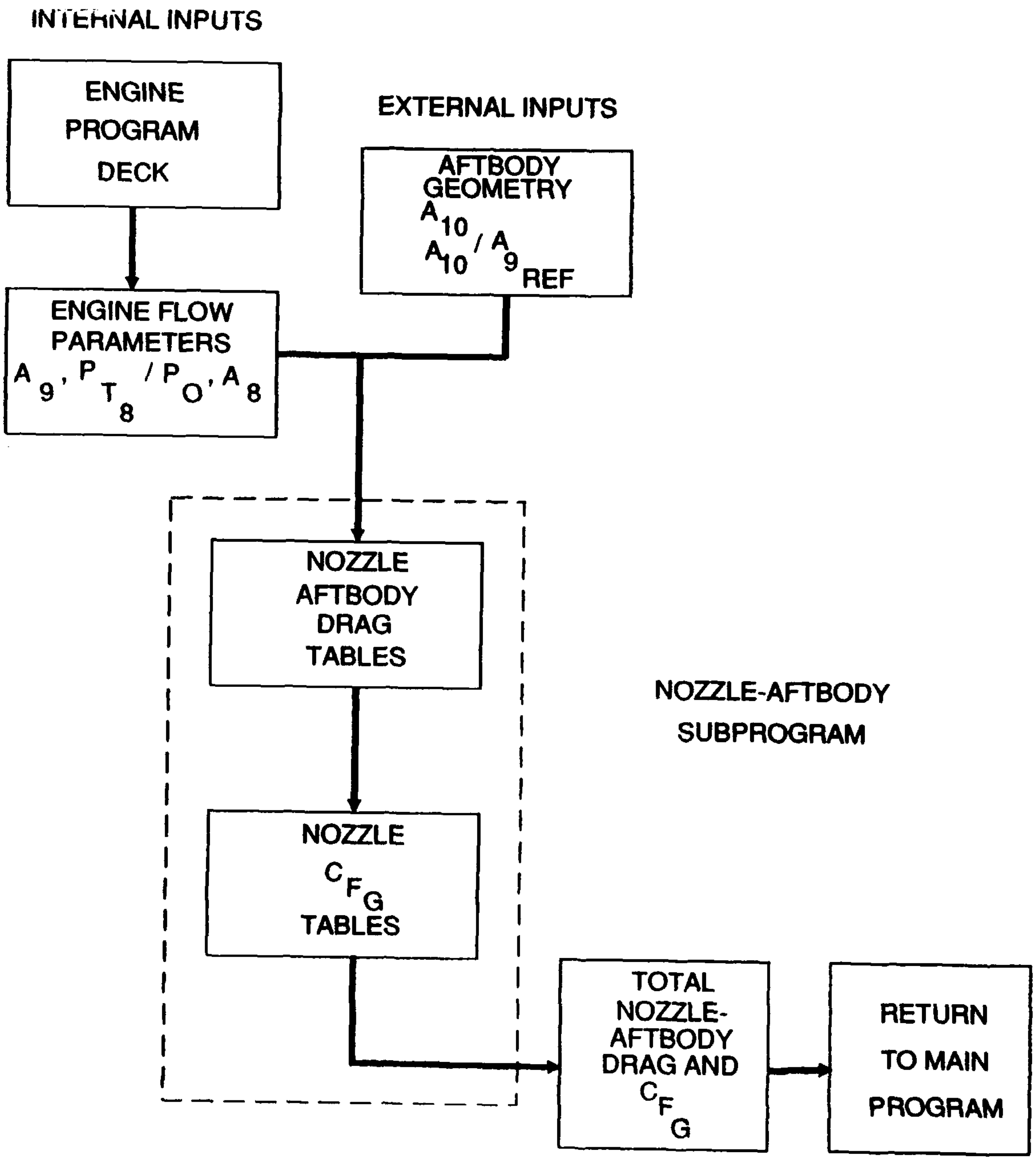


Figure 2.3-7. Nozzle-Aftbody Procedure

The nozzle/aftbody increment to be included in propulsion system installed net thrust will be defined as zero when the nozzle is at its maximum (full-open) geometry and operating at a nozzle static pressure ratio, P_{s9}/P_o, equal to 1.0 (fully expanded). The nozzle/aftbody drag at this condition will be included in the aerodynamic drag. Incremental changes in nozzle aftbody drag due to changes in nozzle/aftbody geometry and/or nozzle static pressure ratio will be included as propulsion system drag. This reference condition is illustrated in Figure 2.3-8 for a typical set of nozzle/aftbody drag data.

THERMODYNAMIC PROPERTIES

Thermodynamic properties required for throat calculations are obtained using the functions shown in Figure 2.3-9. The functions listed here are "curve-fits" of Keenan and Kaye data. The gas tables are primarily used to calculate exhaust nozzle static pressures and jet velocities.

ENERGY BALANCE FOR EXHAUST GAS CALCULATIONS

If the temperature at the engine compressor face, airflow, the bleed mass flow (BL), pressure ratio and fuel flow are known, the exhaust gas enthalpy (h) and relative pressure (P_r) can be calculated from the energy balance:

$$W_2 h_{T_2} + W_f O_{h_B} = W_{18} h_{T_{18}} + W_8 h_{T_8} + W_{BL} h_{T_{BL}}$$

(for either mixed or non mixed flow engines)

For mixed flow fans or a turbojet:

$$W_8 = W_2 - W_{BL} + W_f$$

$$(f/a)_8 = W_f / (W_2 - W_{BL})$$

$$h_{T_8} = (W_2 h_{T_2} + W_f Q h_B) / W_8 \quad (W_{BL} h_{BL} \textit{ is considered negligible})$$

$$P_{r_{T_8}} = f(h_T, f/a)_8$$

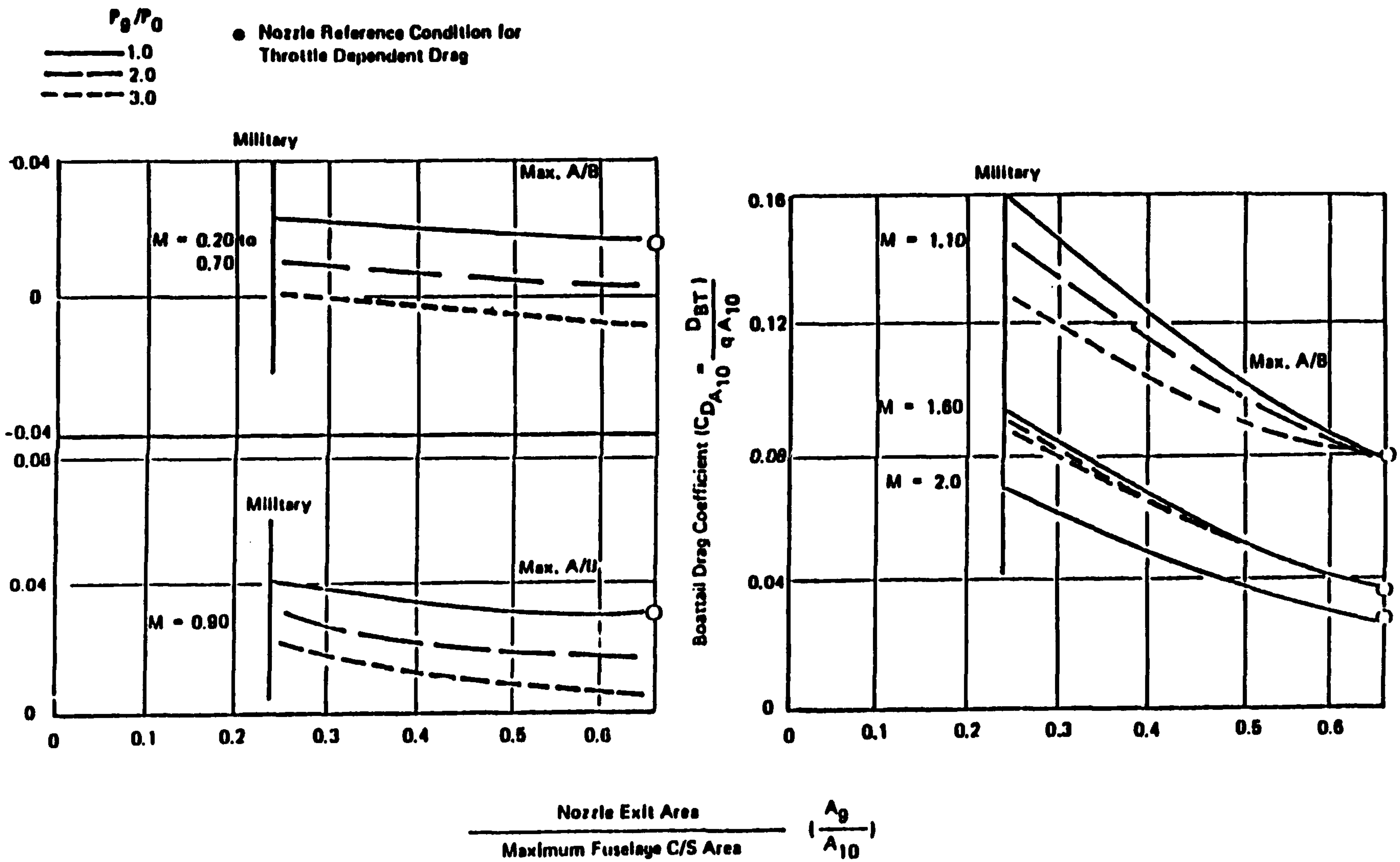


Figure 2.3-8. Typical Nozzle/Aftbody Drag Data

THERMODYNAMIC SUBROUTINE	CALCULATIONS
H = HOFT (T, FOA)	Enthalpy as a function of temperature (degrees R) and fuel-air ratio
T = TOFH (H, FOA)	Temperature as a function of enthalpy and fuel-air ratio
PR = PROFH (H, FOA)	Relative pressure, (P_r) as a function of enthalpy and fuel/air ratio
H = HOFPR (PR, FOA)	Enthalpy as a function of relative pressure and fuel-air ratio
C = COFH (H, FOA)	Sonic velocity as a function of total enthalpy and fuel-air ratio
C = COFHS (H, FOA)	Sonic velocity as a function of static enthalpy and fuel-air ratio

Figure 2.3-9. Thermodynamic Subroutines

NOZZLE GROSS THRUST CALCULATION

The calculation procedure in this section applies to both mixed and non mixed flow nozzles.

CONVERGENT NOZZLE

The velocity at the throat for a convergent nozzle is a function of the total enthalpy (assuming the throat is choked).

$$C_8 = f(h_{T,} f/a)_8$$

and the static pressure is a function of the static enthalpy

$$h_{T_8} = h_8 + \frac{(C_8)^2}{2gJ}$$

$$T_8 = f(h, f/a)_8$$

$$P_{r_8} = f(h, f/a)_8$$

$$P_{T_8}/P_8 = (P_{r_T})_8/P_{r8}$$

P_{T_8} is obtained from the tabulated engine input data as a F(P.S., alt., M) or it is calculated by the procedure described in the "Nozzle Pressure Ratio Calculation" section.

$$P_8 = P_{T_8}/(P_{T_8}/P_8)$$

The area of the throat is

$$A_8* = \left(\frac{WRT}{PC_8}\right)$$

and the thrust is

$$F_g = \frac{W_8 V_8}{g} + A_8*(P_8 - P_{amb})$$

<u>CONVERGENT-DIVERGENT NOZZLE</u> (Fully Expanded)

If the exhaust flow is fully expanded, the static pressure of the nozzle exit is equal to ambient, and the exit velocity is a function of the total to static enthalpy.

$$P_{r_9} = P_{r_8}(P_{amb}/P_8)$$
$$h_9 = f(P_r, f/a)_9$$
$$T_9 = f(h, f/a)_9$$

Since $h_8 = h_9$

$$V_9 = [2gJ(h_{T_8} - h_9)]^{1/2}$$

The exit area is

$$A_9 = W_9 R_8 T_9 / P_{amb} V_9$$

and the gross thrust is

$$F_g = \frac{W_9 V_9 C_{F_G}}{g}$$

CONVERGENT-DIVERGENT NOZZLE (Not Fully Expanded)

If the exit area of a convergent-divergent nozzle is less than required for full expansion, the exit static pressure will be higher than ambient. The throat conditions are known; therefore, a guessed exit velocity gives:

$$h_9 = h_{T_8} - V_9^2/2gJ$$
$$T_9 = f(h, f/a)_9$$
$$P_{r9} = f(h, f/a)_9$$
$$P_9 = \frac{P_{r9}}{P_{r8}}$$
$$W_g = \frac{P_9}{R(T_9)} A_9 V_9 = (PAV)_9 = \frac{(PAV)}{RT} 9$$

(C_S - stream thrust coefficient)

An iteration on V_9 to make $W_9 = W_8$ will result in the exit conditions for a given area.

The gross thrust is:

$$F_g = (\frac{WV}{g} + PA)_9 \; C_S - P_{amb} A_9$$

NOZZLE PRESSURE RATIO CALCULATION

The exhaust nozzle pressure ratio can be calculated if thrust, fuel flow, and airflow are known. The gross thrust is calculated as follows:

$$F_g = (F_{net} + F_{ram}) C_{F_G}$$
$$F_{ram} = \frac{W_2 V}{g}$$

and the nozzle exit conditions are calculated by assuming that flow is fully expanded:

$$
\begin{aligned}
W_8 &= W_2 - W_{BX} + W_f \\
h_{T_8} &= h_{t_2} = h_{T_2} + (Q_B W_f / W_8) \\
T_{T_8} &= f\ (h_T, f/a)_8 \\
V_9 &= F_g(g) / W_8 \\
h_9 &= h_{T_8} - V_9^2 / 2gJ \\
P_{r_9} &= f(h, f/a)_8 \\
(P_{r_T})_8 &= f(h_T, f/a)_8
\end{aligned}
$$

since $P_9 = P_{amb}$

$$P_{T_8}/P_{amb} = (P_{r_T})8/P_{r_9}$$

The pressure ratio calculation will be in error, an amount relative to the value of the thrust coefficient (C_{FG}), because this is usually unknown if pressure ratios and exhaust areas are not given.

2.4 Program Structure

2.4.1 Overview

The PWSIM program system consists of:

(i) a driver program that controls the sequence of computations determined by the input options selected by the user

(ii) a library of propulsion installation routines

(iii) a set of formatted data files containing propulsion installation data for a wide variety of engine installation configurations and operating conditions

(iv) a library of mission performance calculation modules

(v) a set of libraries of baseline geometry, drag and weight scaling routines - one library for each of the baseline configurations

(vi) an input data set defining the user's selection of the various program options and the parameters describing the deviation of design from the baseline. This data set consists of a FORTRAN NAMELIST containing both numerical and character data

(vii) a formatted data set containing definitions of up to 20 missions

(viii) a further set of input that is supplied by the user either interactively (at the computer terminal) or as data entries in the input stream of a batch job.

The complicated nature of engine installation calculation and the large amount of data generated during the installation calculations necessitates that the program be arranged in an overlay format to keep the core memory requirements within acceptable limits. Figure 2.4-1 illustrates the hierarchy of the overlay structure (and also indicates the names of the routines that are accessed within each overlay).

2.4.2 Program Flow

The sequence of activity in the main overlay is shown as a functional flow chart in Figure 2.4-2.

The sequence of calculations, shown in Figure 2.4-2 is as follows:

1. Fetch the General Input and Mission Definition Files to the local operating system.

2. (i) Read and check the MISSION input file
(ii) Read the GENERAL input file (NAMELIST file)

3. (a) If an error is detected or an "end of job" input flag (ENDJOB = 'YES') is read, stop execution;

(b) If all is well proceed to Step 4.

4. Execute the baseline geometry calculations to evaluate:

engine scale

nozzle area

aftbody drag reference area

These data are required for subsequent engine installation calculations.

5. (a) If installed engine data are already available proceed to Step 9 (this is denoted by the flag ENGRED set to 'YES')

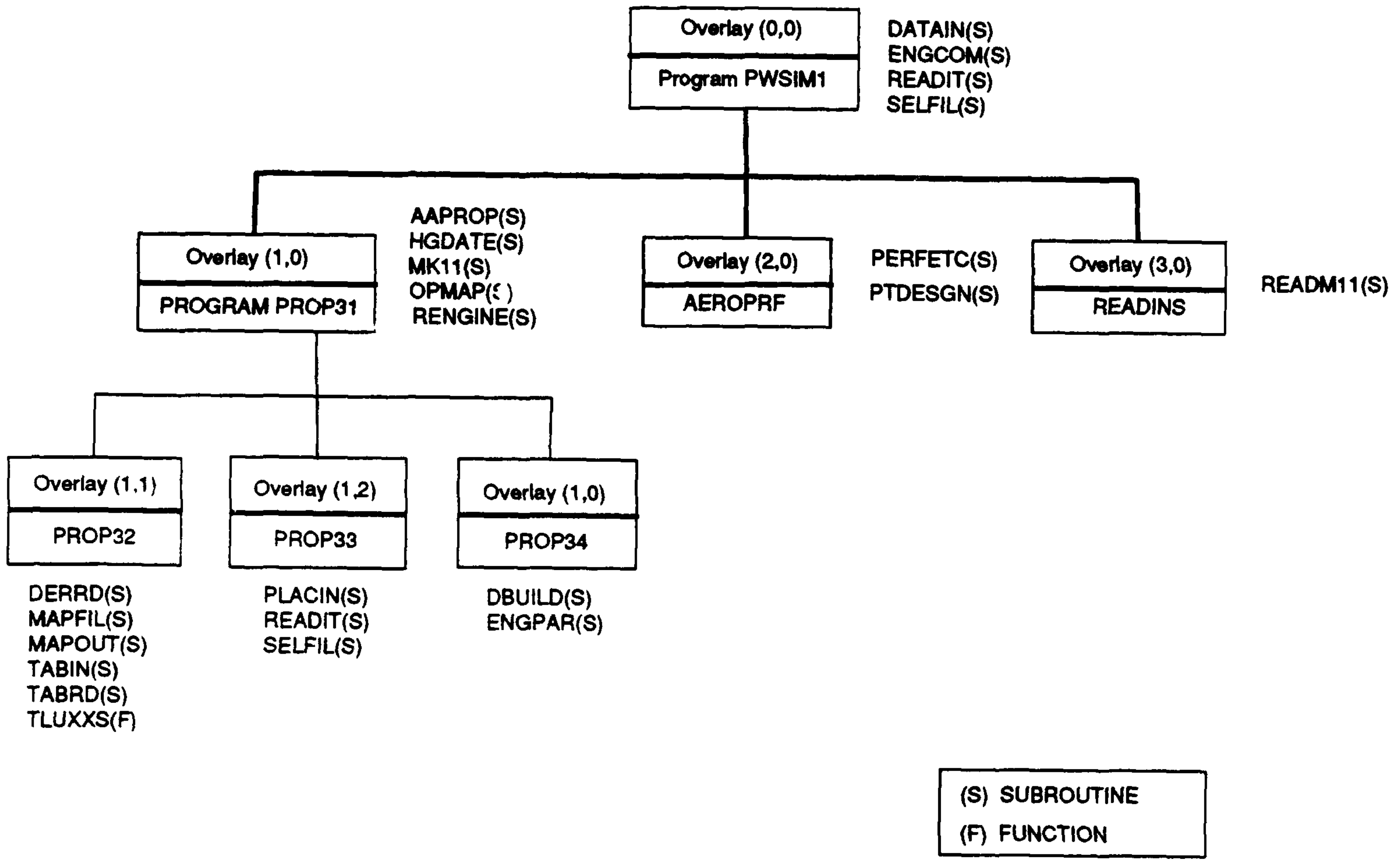


Figure 2.4-1. Overlay Hierarchy

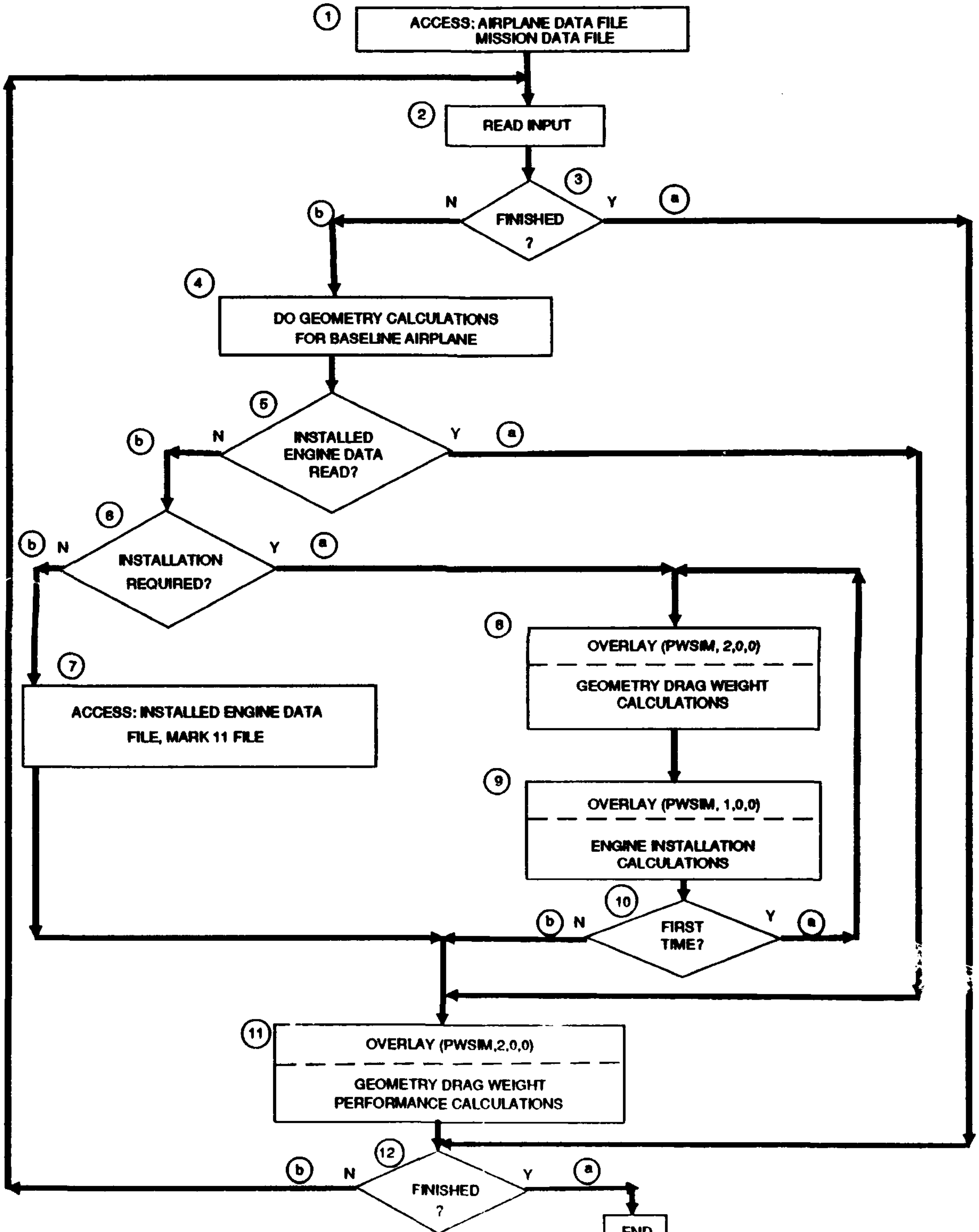


Figure 2.4-2. Functional Flowchart of PWSIM

(b) If no engine data have been accessed proceed to Step 6. This is detected by the flag ENGRED set to a value of 'NO'.

6. (a) If the engine installation procedure is to be used (if INSTREQ = 'YES') go to Step 8.

(b) If the engine data is to be read from a file of already installed engine performance - hereafter referred to as a "Mark 11" file - (if ENGRED = 'NO') go to Step 7.

7. Access the specified Mark 11 installed engine data file and read its contents into memory. Set the flag that indicates that engine performance data has been read (ENGRED = 'YES').

8. Perform the geometry calculations needed to calculate the aftbody drag reference area, A10, as this quantity is needed for the installation calculations. On the first pass through the geometry calculations, the airplane designer's guess at the inlet capture area may be used for the geometry calculations. The correct value is obtained from the installation calculations which cannot be performed until A10 is known. Thus, a two-step iteration is necessary. The first iteration calculates the A10 value using an assumed capture area and then recalculates the capture area using the new A10.

9. Perform the engine installation calculations using the current value of A10, calculate the correct value of reference capture area, RACAPT. If this is the second time through this block:

write results to TAPE20

set flap to show installation is complete (INSTREQ = 'NO'

set flap to show engine data has been read (ENGRED = 'YES').

10. Check to see if the installation calculation has already been accessed.

a. Return to geometry calculation with correct value of RACAPT

b. Go on to Step 11.

11. Perform the airframe technology calculations (geometry, drag and weights) and the airplane mission calculations. Write the results to TAPE 6.

12. Check the status of the 'end of job flag' ENDJOB

    a. If ENDJOB = 'YES' stop execution

    b. If ENDJOB = 'NO' go to Stop 2-(ii).

It is to be noted that the engine performance data are accessed (by reading the MARK 11 file or by installation) only once per job. If subsequently the airplane size (and thus engine size) is changed (for example, during a sizing iteration), then the installed engine performance data are scaled rather than performing another installation. To be strict, the uninstalled data should be scaled prior to installation; since airplane sizing can involve several iterations of engine size, the technique of scaling the installed data is used to keep computation time as low as possible.

An alternative way of looking at the program overall structure is shown in Figure 2.4-3. This shows the main program module and the three subservient libraries. The library shown inside the box of dashed lines is that which contains the geometry, drag, and weight modules for the configuration being studied.

3.0 Data Base Descriptions

Seven weapon system preliminary designs are provided to serve as point-of-departure baseline configurations for the PWSIM program.

The conceptual data bases produced for each configuration consist of several items that, taken together, will give a thorough definition of the design-point aircraft and provide a sound basis for parametric studies. The items contained in each data base are:

(a) an outboard profile, three-view engineering drawing

(b) engineering description

(c) geometric summary

(d) weight statement

(e) drag polars

(f) engine performance data

(g) airplane performance in the design mission

(h) limitations on the applicability of the data base.

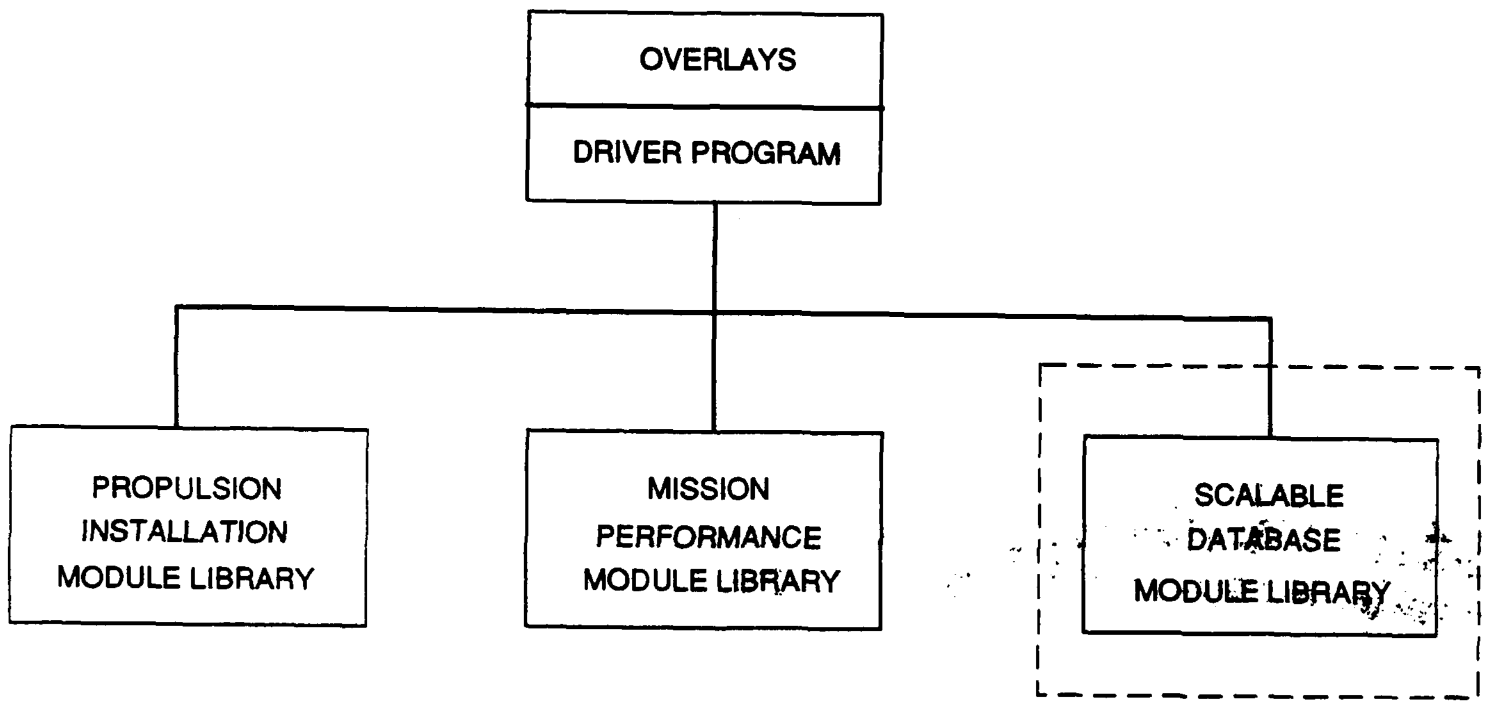


Figure 2.4-3. PWSIM Library Structure

The following sections contain detailed descriptions and comprehensive data summaries of the selected baseline aircraft configurations. Subsequent sections contain representative data allowing the evaluation of the drag and weight penalties of externally carried stores.

3.1 Tactical Fighter - Model 985-420

3.1.1 Concept Description

This aircraft is shown in Figure 3-1. The overall airplane length is 62 ft 2 in and the wing span 50 ft 9 in. The wing has a leading edge sweep of 37.5°, a reference wing area of 571 ft^2, an aspect ratio of 4.5, and a wing thickness ratio of 5% at the side of body and 4% at the tip. A smooth variable camber leading edge is used with a hinged, single slotted trailing edge flap during landing approach. Wing camber is varied automatically throughout the flight envelope for improved lift/drag ratio. Hardpoints are provided for carrying external fuel tanks (for extended range and ferry missions) and alternate weapon configurations.

The airplane is designed for a one-man crew. Located forward, aft, and below the crew compartment are the avionics/electronics equipment compartments. Included in the 1859 lb of avionics equipment are target acquisition, communication, navigation and identification, information managements, and defense functions. ECS equipment, oxygen, and electrical/hydraulic subsystem equipment are located in the fuselage aft of the pilot. The body fuel is carried in integral tanks with a capacity of 12,000 lb of JP-4 fuel.

Two vertical fins are integral with the aft fuselage side walls and have a total area of 110 ft^2. Each uses a conventional rudder (32% of the fin chord). All-moving, slab canards with an exposed total area of 78 ft^2 are used for longitudinal and roll control throughout the entire speed regime. Wing flaperons will augment roll control throughout the flight envelope.

3.1.2 Aerodynamics

Estimated aerodynamic characteristics are presented in this section for the 985-420. Figure 3-2 illustrates the complete drag polar at three key conditions in the flight envelope.

3.1.3 Weights

The weight statement for the 985-420 is shown in Figure 3-3. Weight estimating ground rules and assumptions are:

- MODEL 985-420
- TOGW = 40,000 LBS
- W/S = 70 LBS/FT^2
- S = 571 FT^2
- Λ_{LE} = 37.5
- AR = 4.5
- T/W = 1.25
- T_{SLS} = 25,000 EACH
- EXT. FUEL = 50% FIF

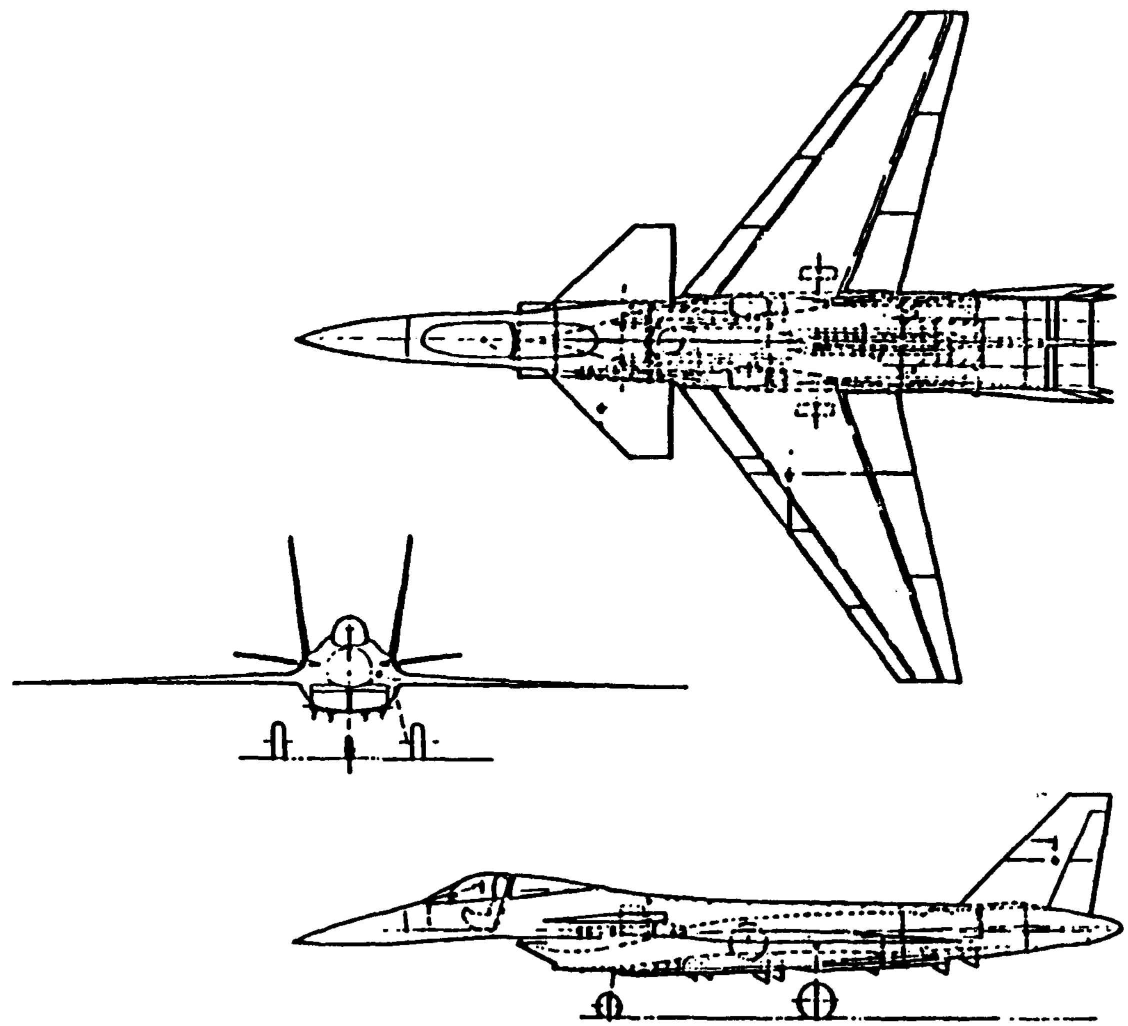


Figure 3-1. Tactical Fighter

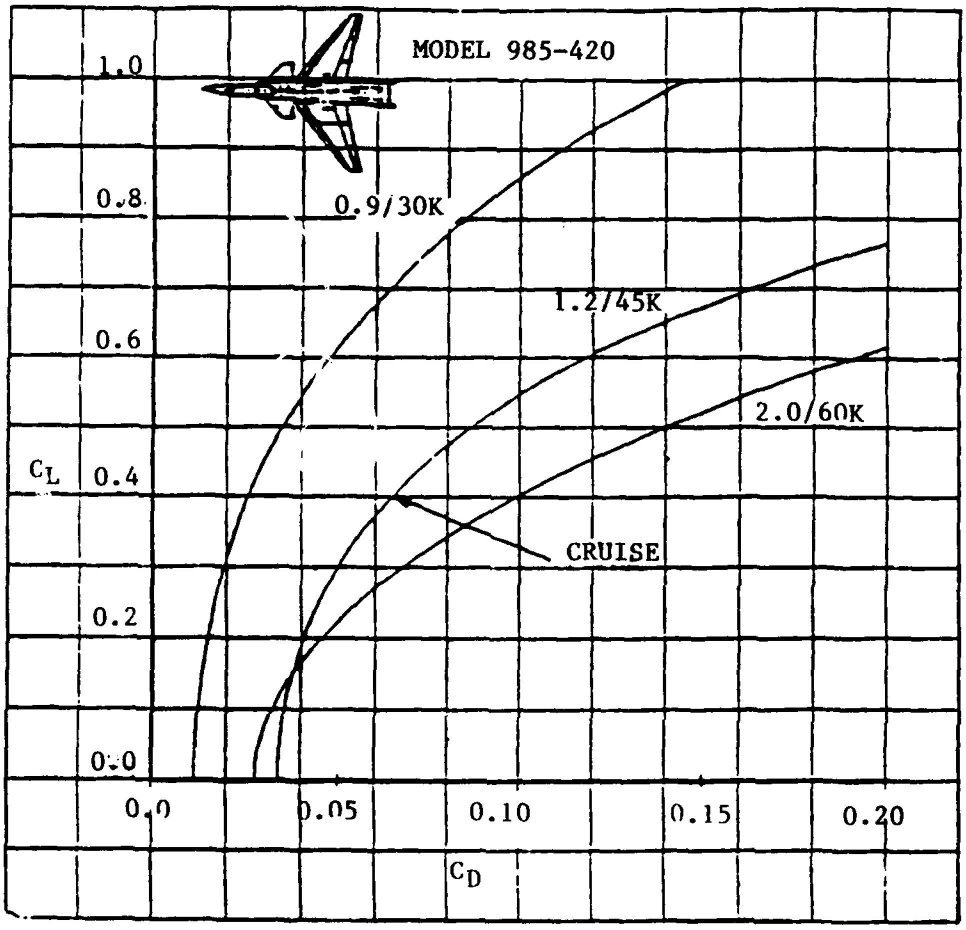


Figure 3-2. Tactical Fighter, Drag Polars

985-42 GROUP WEIGHT STATEMENT PDWTS 01-OCT-82 VERSION 3-MAY-83	WEIGHT-LBS	NOSE STATION 0. WING MAC 151. LEMAC 407. BODY LENGTH 746.	
		Body St	Percent MAC
Wing	3728.	483.	
Canard	566.	295.	
Vertical Tail	665.	678.	
Body	5180.	411.	
Alighting Gear	1541.	410.	
Nacelle or Eng Section	188.	603.	
Air Induction System	963.	341.	
Total Structure	12836.	438.	
Engine + Accessories	4960.	603.	
Starting + Control	160.	387.	
Fuel System	709.	420.	
Total Propulsion	5829.	575.	
Flight Control	939.	521.	
Auxiliary Power Plant	240.	580.	
Instruments	160.	175.	
Hydraulic + Pneumatic	418.	517.	
Electrical	922.	434.	
Avionics	1859.	210.	
Armament	340.	400.	
Furnishings + Equip	220.	170.	
Air Cond + Anti-Icing	756.	435.	
Load + Handling	10.	430.	
Total Fixed Equipment	5864.	370.	
Weight Empty	24528.	454.	30.9
Crew	230.	170.	
Unusable Fuel	130.	420.	
Oil + Trapped Oil	190.	603.	
Gun Installation + Ammo	685.	320.	
Crew Equipment	50.	170.	
AMRAAM Ejectors (6)	390.	450.	
Non-Exp Useful Load	1675.	365.	
Operating Weight	26203.	448.	27.1
Payload	2000.	450.	
Fuel	11797.	420.	
GROSS WEIGHT	40000.	440.	21.6

Figure 3-3. Tactical Fighter Weight Statement

- Fuel tanks are inerted with nitrogen gas from N_2 generators.

- Inflight refueling provisions have been incorporated.

- Arresting tail hooks have not been incorporated.

- Engine inlets, pitot tubes and canopy have de-icing systems.

- Air conditioning systems are closed loop bootstrap plus liquid cycle cooling variety.

- Provisions for weapon hardpoints on wing and fuselage have been incorporated. Each hardpoint assumes multi weapons control in terms of attachment and launching.

- The APU is an IPU "Integrated Power Unit." Its function is to operate as a starter and an emergency power system.

- The landing gear CBR is 9. Note: CBR (California Bearing Ratio) is a measure of the bearing strength of the airfield from which the aircraft must operate.

- Hydraulic system operates at a pressure of 4000 psi.

- Flight control system utilizes fly-by-wire technology.

- Flight design weight equals gross weight less 20 percent of the on-board fuel weight.

- Landing weight equals the gross weight less 40 percent of the on-board fuel weight.

- No weight penalty has been assessed for incorporation of Mission Adaptive Wing (MAW).

- TAD is 1987, IOC is 1993.

3.1.4 Propulsion System

Uninstalled engine data were computed using the PWA engine cycle program, PWA CCD 1178-06.01. This engine has a bypass ratio of 1.0, an operating pressure ratio of 25, and a max burner temperature of 3000°F. Installation effects were estimated using the Engine Installation Analysis Program. The inlets are under wing, centerline mounted, two-dimensional external compression downward spilling fixed horizontal ramp, which allow the airplane to achieve the M = 2.0 dash speed at altitude, and provide inlet flow protection during high angle-of-attack maneuver conditions. The inlet ducts are designed with structural radar-absorbent materials (RAM). The capture area of each inlet is 5 ft^2.

Engine mounted 2-D, C-D nozzles are arranged side-by-side and incorporate variable throat area capability for augmented engine operation. Adequate cooling flow is provided to nozzle surfaces to minimize IR signature and to allow application of RAM for reduced RCS. Installed thrust and SFC curves for subsonic and supersonic cruise conditions are shown in Figure 3-4 through 3-5.

3.1.5 Performance

This aircraft was designed to fly to a radius of 1000 nmi and patrol on station for 1 hour before returning to the starting point. A summary of the basic sizing mission is shown in Figure 3-6. A summary of the design mission segment by segment is given in Figure 3-7.

3.2 Supersonic Interceptor - Model 985-430

3.2.1 Concept Description

This vehicle, illustrated in Figure 3-8 has an overall airplane length of 93 ft 4 in and a wing span of 38 ft 5 in. The wing has a leading edge sweep of 75° on the main inner wing section and 55° on the outboard section, a reference wing area of 1002 ft^2, an aspect ratio of 1.47, and a wing thickness ratio of 4.4% at the side of body and 1.9% at the tip. A smooth variable camber leading edge is used with a hinged, single slotted trailing edge flap during landing approach. Wing camber is varied automatically throughout the flight envelope for improved lift/drag ratio. At low speeds, the leading edge vortex flap is deployed, as is the high lift canard. The wing provides volume for approximately 7770 lb of fuel in integral wing tanks.

The airplane is designed for a one-man crew. Located forward, aft, and below the crew compartment are the avionics/electronics equipment compartments. Included in the 2699 lb of avionics equipment are target acquisition, communication, navigation and identification, information management, and defense functions. ECS equipment, oxygen, and electrical/hydraulic subsystem equipment are located in the fuselage aft of the pilot. The body fuel is carried in integral tanks with a capacity of 18,130 lb of JP-4 fuel.

The vertical fin has an area of 130 ft^2. A conventional 30% rudder is incorporated. Additional directional stability is provided by a ventral fin.

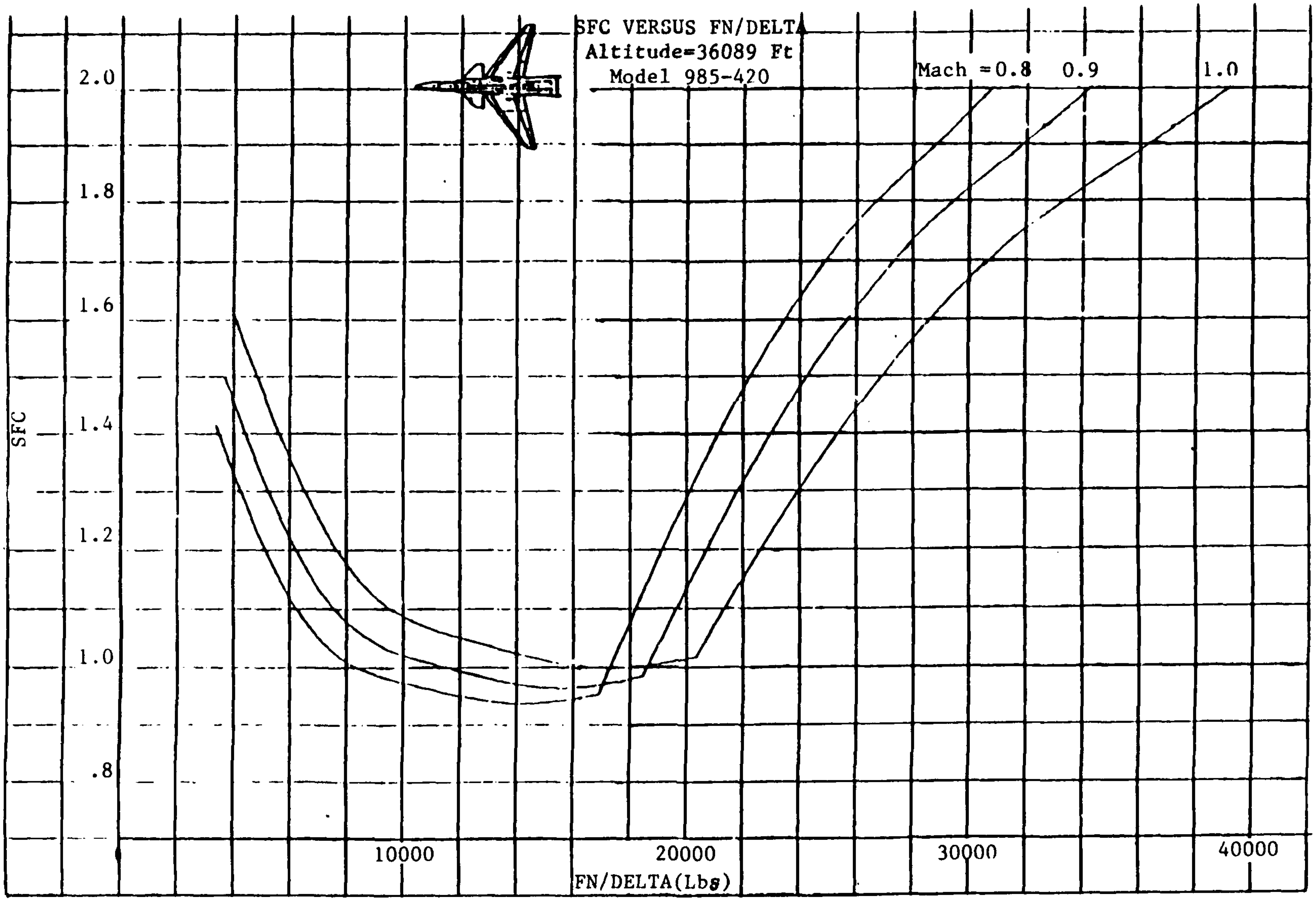


Figure 3-4. Tactical Fighter Subsonic Cruise SFC

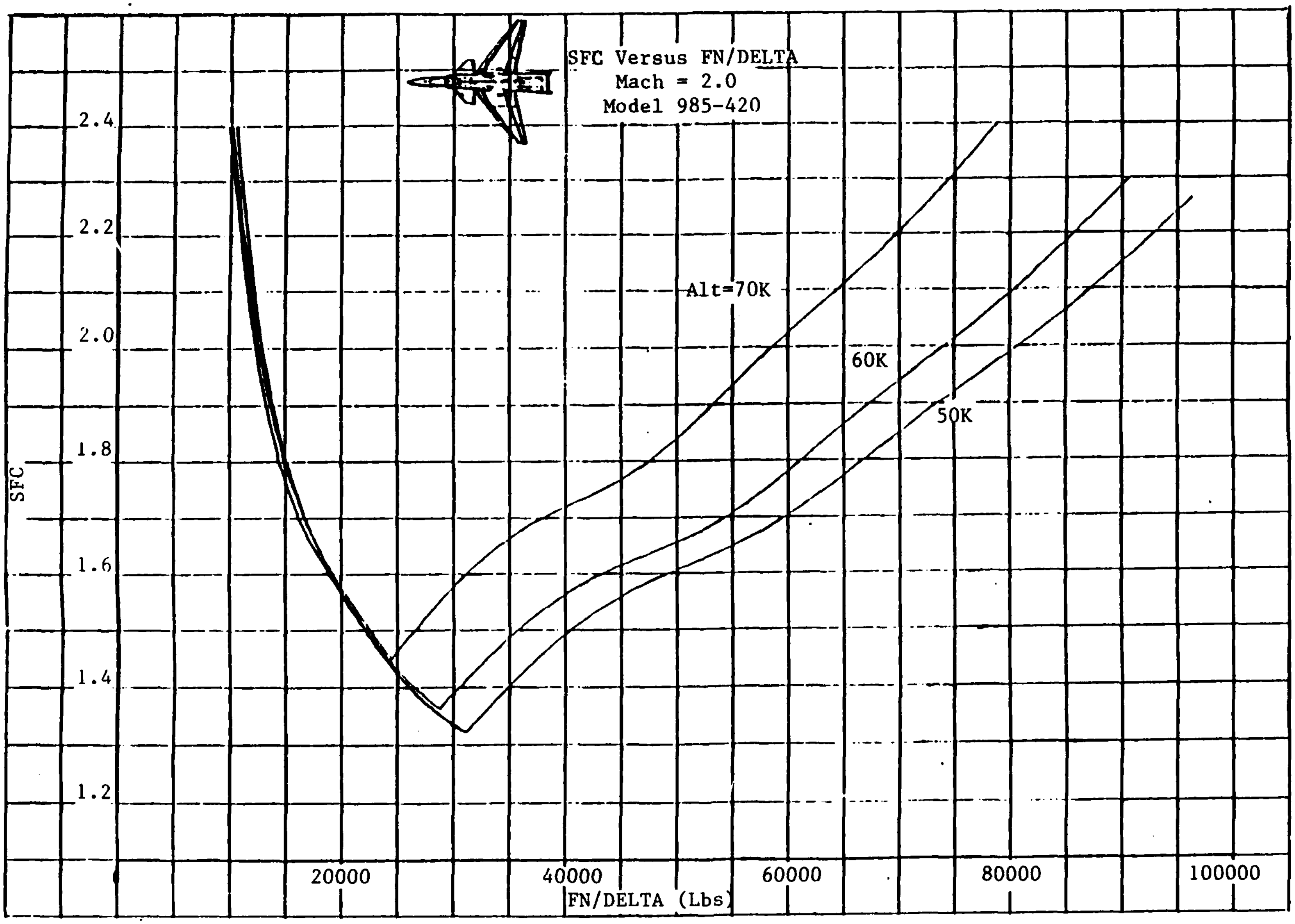


Figure 3-5. Tactical Fighter Supersonic Cruise SFC

1. TAKEOFF FUEL ALLOWANCE
    - 2.5 MIN IDLE FUEL FLOW
    - 1/2 MIN MAX POWER FUEL FLOW
    - MAX POWER ACCELERATION TO CLIMB SPEED
2. MIL POWER CLIMB
3. SUBSONIC CRUISE; OPTIMUM MACH/ALTITUDE
4. LOITER ON STATION; OPTIMUM MACH/ALTITUDE
5. COMBAT
    - (1) MA POWER TURN; M=.80/ALTITUDE=10,000 FT
    - RELEASE PAYLOAD
6. ENROUTE CLIMB
7. SUBSONIC RETURN TO BASE; OPTIMUM MACH/ALT' 'JDE
8. RESERVES; 20 MIN SEA LEVEL LOITER, OPTIMUM MACH NUMBER

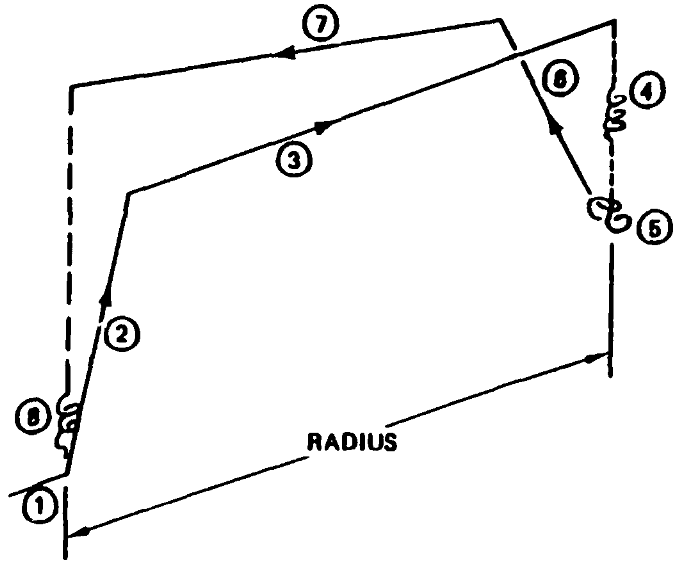


Figure 3-6. Ta:tical Fighter Design Mission Profile

MISSION SEGMENT	MACH	ALTITUDE (FT)	DISTANCE (NMI)	FUEL (LB)	L/D	SFC
TAKEOFF/ACCEL	0	0	0	1700		
CLIMB	0 to 0.75	43,100	33	800		
CRUISE	0.86	43,100	987	6030	14.5	1.07
LOITER	0.69	36,200	0	2550	15.5	1.07
COMBAT	0.80	10,000	0	500	8.2	2.33
CLIMB	0.8 to 0.86	50,000	31	540		
CRUISE	0.86	50,000	989	4560	14.4	1.09
LOITER	0.25	0	0	1040	16.4	1.85
TOTAL			2040	17,720		

Figure 3-7. Tactical Fighter Design Mission Summary

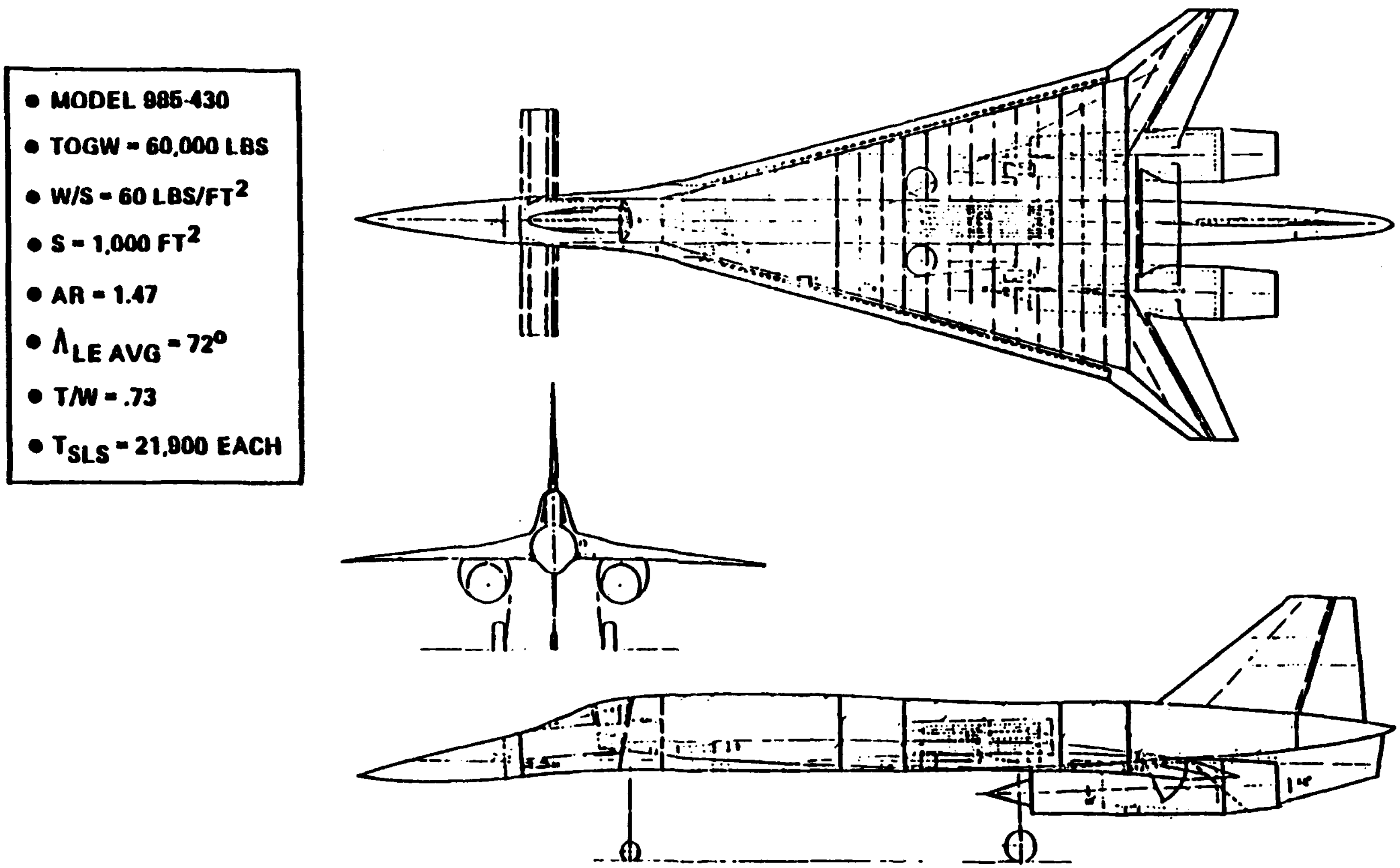


Figure 3-8. Supersonic Interceptor

The stowable canard has an area of 52 ft^2 and provides a high lift capability at low speed through the use of a slat/double slotted flap airfoil. Roll control is provided by wing spoiler/slot deflectors and trailing edge flaperons.

3.2.2 Aerodynamics

Estimated aerodynamic characteristics are presented in this section. Drag polars for the critical mission Mach numbers are shown in Figure 3-9.

3.2.3 Weights

The weight statement and weight related design data are tabulated in Figure 3-10. Weight estimating ground rules are the same as those applicable to the tactical fighter (985-420) and are listed in Section 3.1.3.

3.2.4 Propulsion System

The design mission for the Model 985-430 requires an engine that can operate with low specific fuel consumption during cruise at Mach 3.0 and an altitude of 70,000 feet. To meet this goal, engines of bypass ratio of 0.2, overall pressure ratio of 10 and a maximum burner temperature of 3000°F were selected.

Two General Electric Mach 3.0 advanced technology afterburning (GE16/J6-B1) and dry (GE16/J5-H3R) turbojets provide the necessary propulsion. The inlets are under wing, axisymmetric mixed compression, which allow the airplane to achieve the M 3.0 combat speed at altitude, and provide favorable interference with the wing. The inlet ducts are designed with structural radar-absorbent materials (RAM). The capture area of each inlet is 10.2 ft^2. Engine mounted axisymmetric nozzles incorporate variable throat area capability for augmented engine operation.

Installed SFC data are presented in Figures 3-11 and 3-12.

3.2.5 Performance

The Supersonic Interceptor has been designed to fly a 1000-nautical-mile radius intercept mission out and back at Mach 3.0 (Figure 3-13).

A summary of the design mission history is shown in Figure 3-14.

3.3 Supersonic Intercontinental Cruise Missile

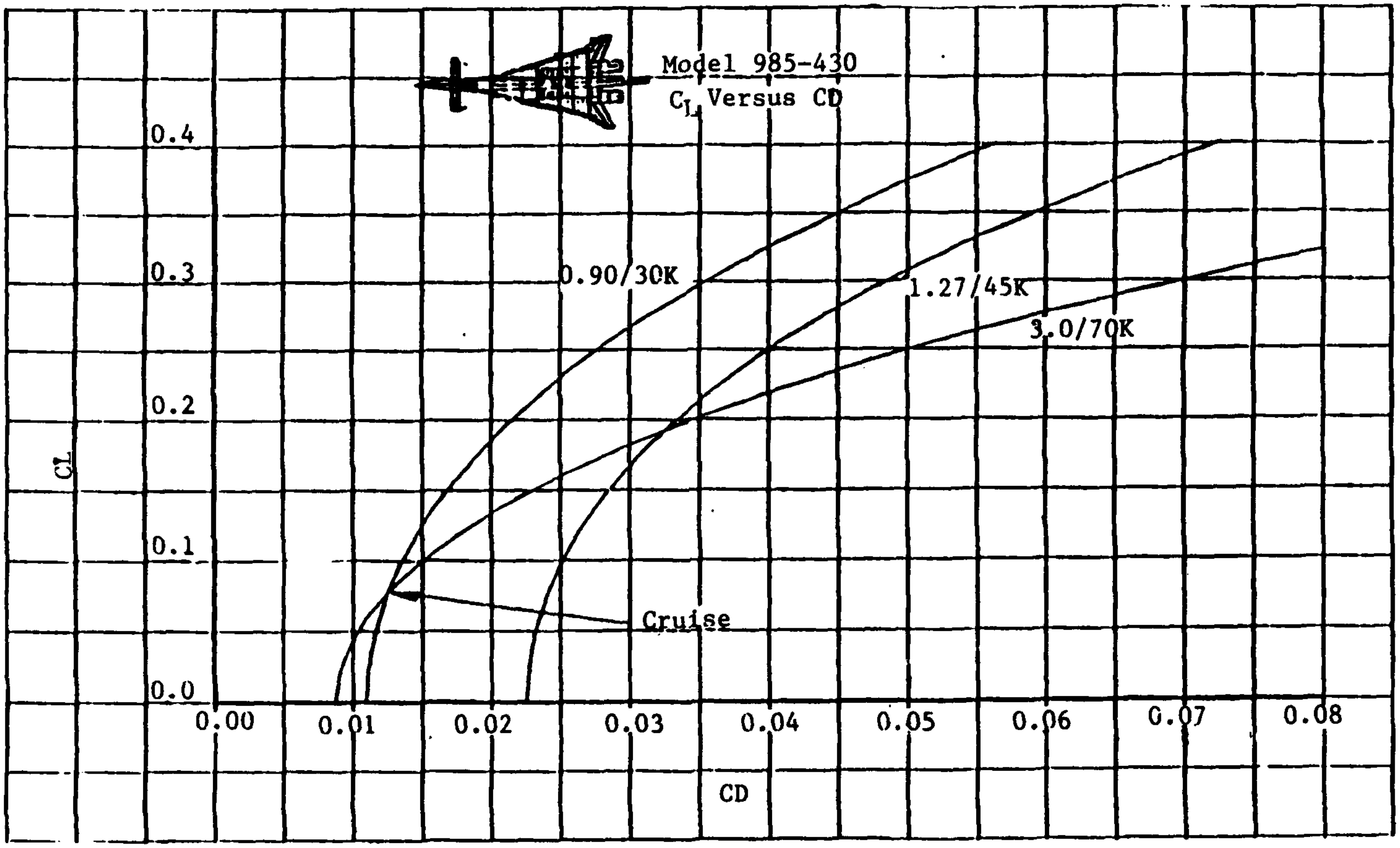


Figure 3-9. Supersonic Interceptor Drag Polars

985-430 INTERCEPTOR GROUP WEIGHT STATEMENT PDWTS 01-OCT-82 VERSION 9-MAY-83	WEIGHT-LBS	NOSE STATION 0. IN WING MAC 376. IN LEMAC 503. IN BODY LENGTH 1120. IN	
		Body Sta	Percent MAC
Wing	5420.	678.	
Canard	220.	197.	
Vertical Tail	801.	980.	
Body	4559.	504.	
Alighting Gear	2422.	597.	
Nacelle or Eng Section	703.	865.	
Air Inducting System	483.	768.	
Total Structure	14607.	631.	
Engine + Accessories	6342.	865.	
Starting + Control	150.	768.	
Fuel System	949.	670.	
Total Propulsion	7441.	838.	
Flight Control	1075.	785.	
Auxiliary Power Plant	240.	830.	
Instruments	160.	285.	
Hydraulic + Pneumatic	831.	753.	
Electrical	1080.	639.	
Avionics	2639.	380.	
Armament	340.	460.	
Furnishings + Equip	315.	280.	
Air Cond + Anti-Icing	1718.	641.	
Load + Handling	10.	640.	
Total Fixed Equipment	8468.	565.	
Weight Empty	30516.	663.	42.6
Crew	230.	280.	
Unusable Fuel	259.	670.	
Oil + Trapped Oil	171.	865.	
Gun Installation + Ammo	685.	390.	
Crew Equipment	50.	280.	
AMRAAM Ejectors (6)	390.	680.	
Rotary Rack	300.	680.	
Non-Exp Useful Load	2085.	545.	
Operating Weight	32601.	656.	40.6
Payload	2000.	680.	
Fuel	25399.	670.	
GROSS WEIGHT	60000.	663.	42.4

Figure 3-10. Supersonic Interceptor Weight Statement

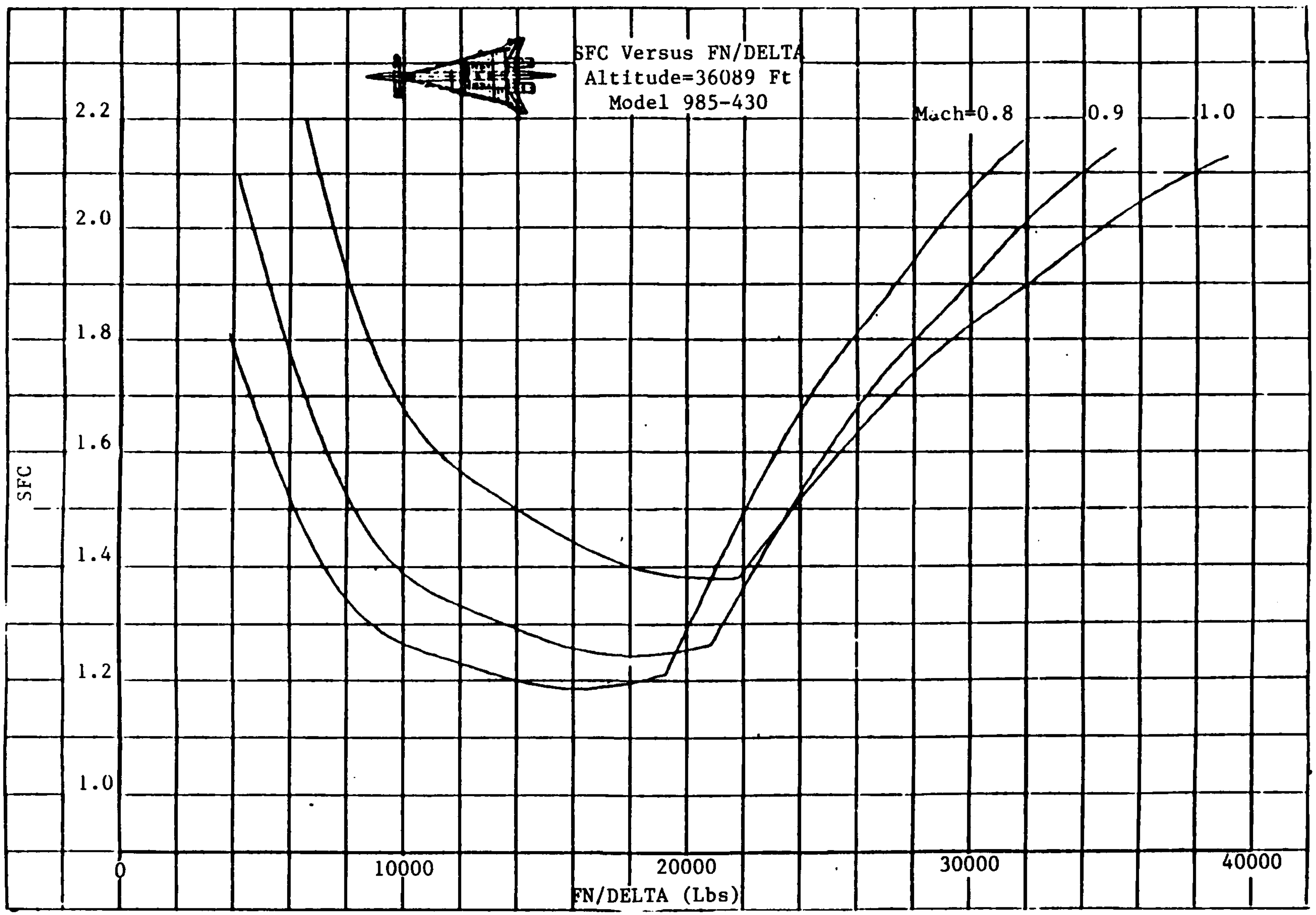


Figure 3-11. Supersonic Interceptor Subsonic Cruise SFC

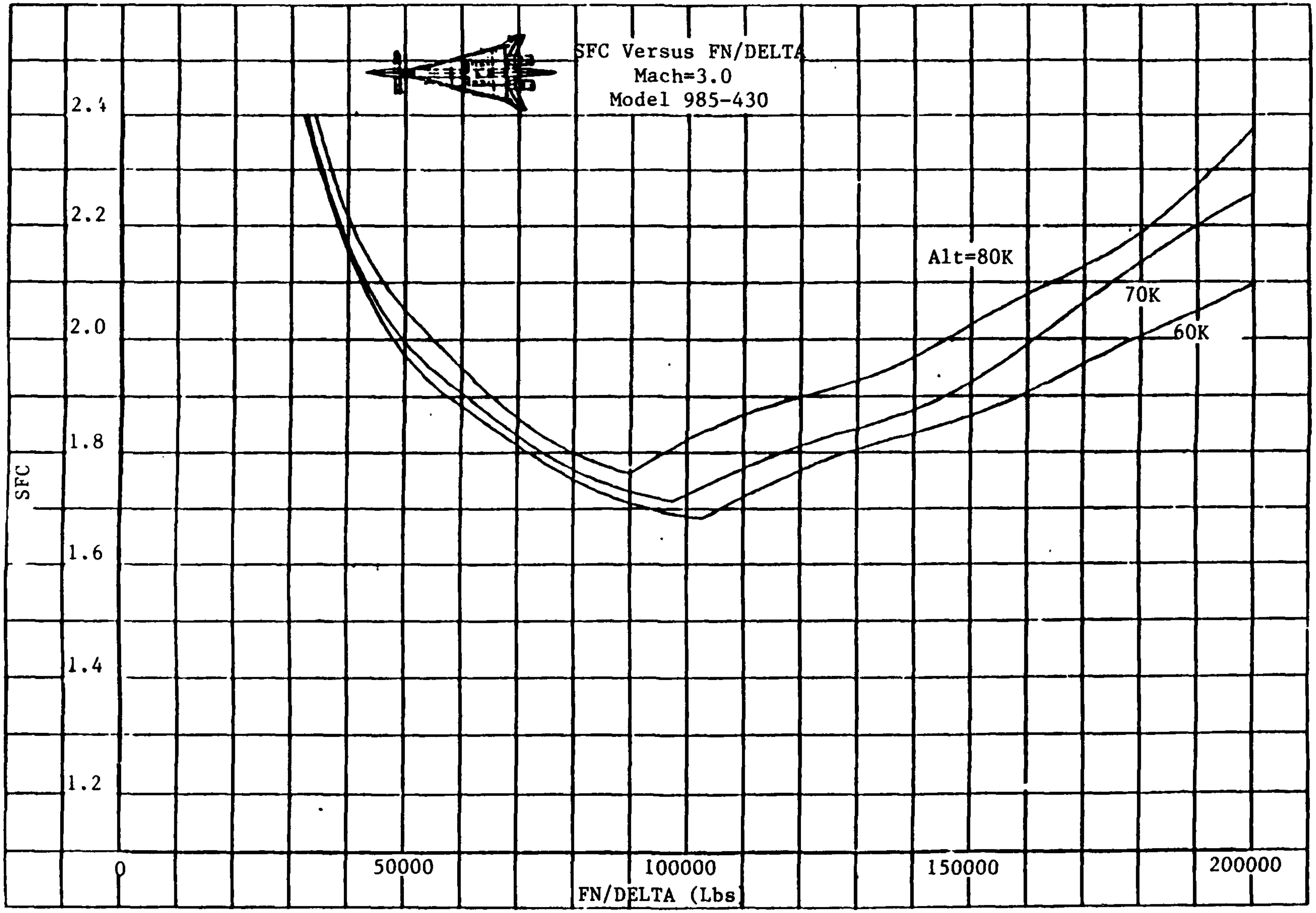


Figure 3-12. Supersonic Interceptor, Supersonic Cruise SFC

1 TAKEOFF FUEL ALLOWANCE
- 2.5 MIN IDLE FUEL FLOW
- 1/2 MIN MAX POWER FUEL FLOW
- MAX POWER ACCELERATION TO CLIMB SPEED

2 MAX POWER CLIMB

3 SUPERSONIC CRUISE TO INTERCEPT; OPTIMUM ALTITUDE

4 COMBAT
- (1) MAX POWER TURN
- RELEASE PAYLOAD

5 SUPERSONIC CRUISE RETURN TO BASE

6 DECEL/DESCENT TO SEA LEVEL

7 RESERVES; 20 MIN SEA LEVEL LOITER, OPTIMUM MACH NUMBER

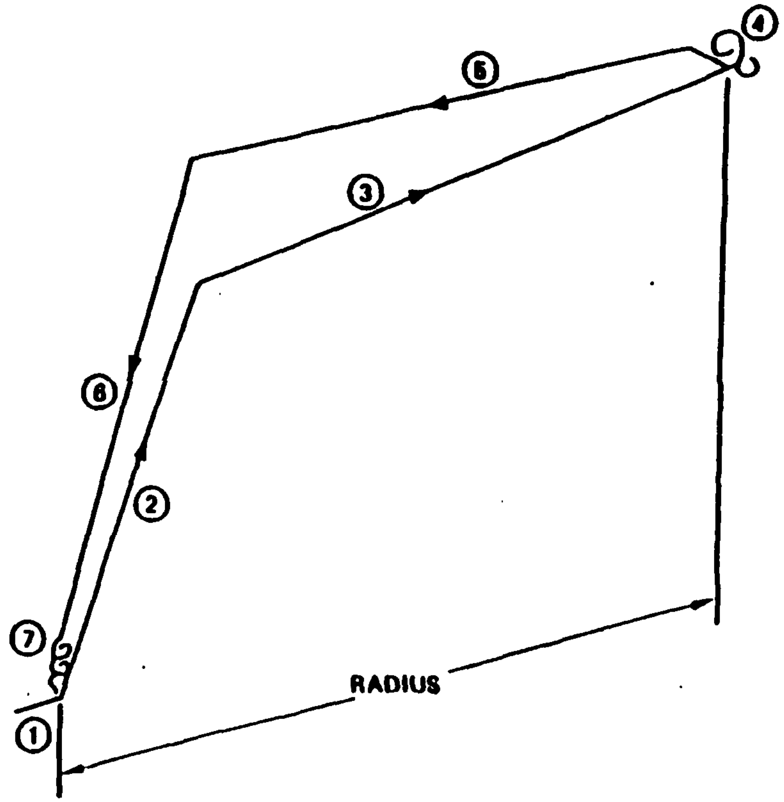


Figure 3-13. Supersonic Interceptor Mission Profile — Supersonic Out and Return

MISSION SEGMENT	MACH	ALTITUDE (FT)	DISTANCE (NMI)	FUEL (LB)	L/D	SFC
TAKEOFF/ACCEL	0	0	0	1800		
CLIMB	0 to 3.0	69,500	66	6200		
CRUISE	3.0	69,500	924	7030	6.33	1.73
COMBAT	3.0	67,500	0	3000	5.8	2.34
CRUISE	3.0	75,000	832	5000	6.2	1.74
DECEL/DESENT	3.0 to .85	0	158	520		
LOITER	.33	0	0	1730	11.1	1.72
TOTAL			1980	25280		

Figure 3-14. Supersonic Interceptor Design Mission Summary

3.3.1 Concept Description

An outboard profile of this vehicle is shown in Figure 3-15. The overall vehicle length is 30 feet and the wing span is 15.5 feet. The wing has a leading edge sweep back of 70 degrees and a trailing edge sweep forward of 40 degrees. The wing has a constant thickness/chord ratio of 0.03 and a taper ratio of zero (except that rounding the wing tips preserves a finite material thickness at the tip).

The design is sized to achieve a range of approximately 3500 nautical miles using present state-of-the-art turbojet propulsion and JP-10 type fuel. The payload consists of a single ballistic vehicle having a suitable yield. The ballistic vehicle is shaped and treated with radar absorbing material for penetration of the terminal defenses in the target area. The avionics system incorporates an inertial system having the capability to receive updates from a Star Tracker or GPS thereby achieving the required terminal accuracy.

The reference midcourse cruise configuration is designed for air launch from a carrier aircraft at a Mach number of 0.6 at 30,000 feet altitude. Solid rocket boosters take the missile from air carrier loiter conditions of M = 0.6 and 30,000-foot altitude to cruise Mach number and altitude of 3.5 and 85,000 ft, respectively. Two boost motors, one on each side of the lower surface of the fuselage/wing intersection, are used to minimize overall carriage length.

Insulated structure is employed with the insulation protected by a thin outer layer of titanium having a high emissivity coating. The load carrying structure is Epoxy-Graphite for those parts of the structure where temperatures do not exceed 400°F or Polyimide Graphite (up to 600°). The min k insulation passively cools the fuel so that a 350° condition (with 35 PSIA vapor pressure) is not exceeded for the flight. Launch is assumed at high altitude, -65°F condition. The warhead is also passively cooled. An allowance for active cooling of the electronics is included in the fixed equipment.

3.3.2 Aerodynamics

The drag of this configuration has been estimated using the results of wind tunnel tests carried out in the NASA Ames 2 x 2-foot transonic and 10 x 14-inch supersonic wind tunnels on a model approximating this configuration. The model differed from the current design in that:

- o the tested model had a semicircular, underwing fuselage

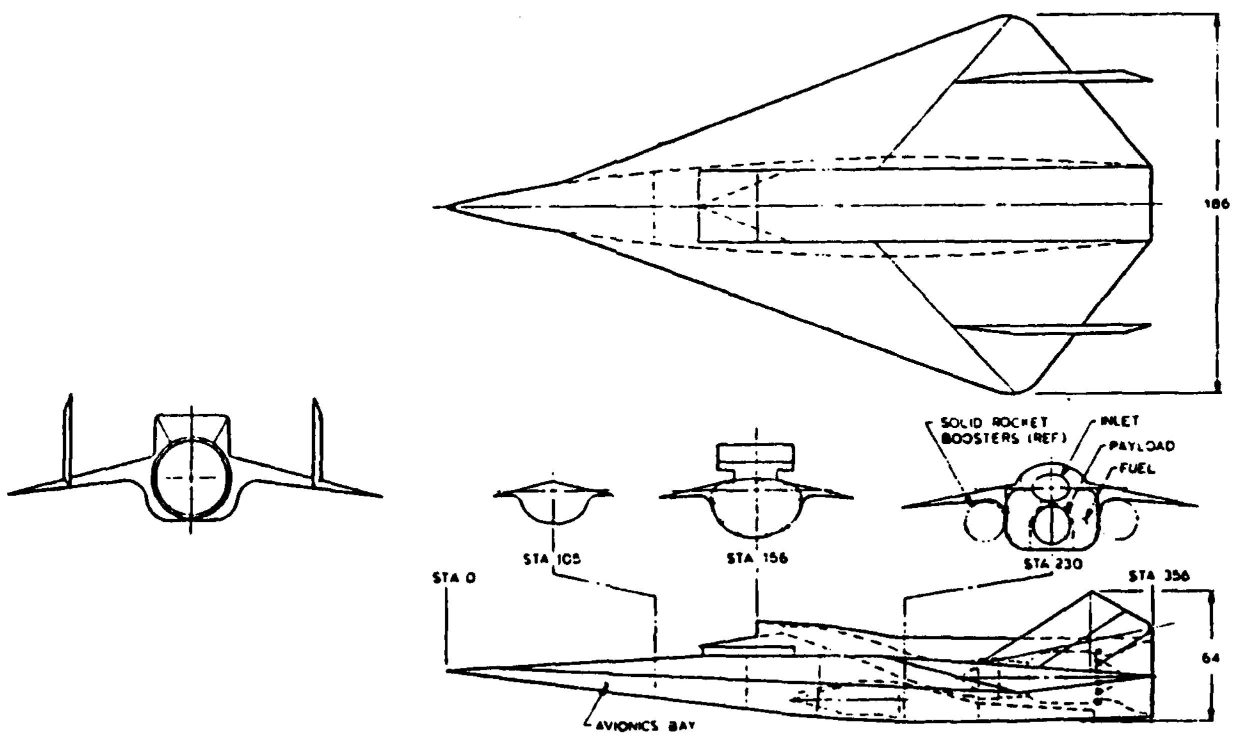


Figure 3-15. Supersonic Cruise Missile

- o the tested model had a rearward sweep wing trailing edge
- o the tested model had no engine installation
- o the tested model had one centrally located vertical fin whereas the current design has two fins mounted at the 65% spanwise station on the wings
- o the test Reynolds Number at M = 3.5 was 5.9 million compared with 22.0 million for the full-scale vehicle.

Appropriate corrections to the test data result in the drag polars shown in Figure 3-16.

3.3.3 Weights

The weight statement for the baseline Supersonic Cruise Missile is shown in Figure 3-17.

The weight calculations take due account of the design peculiarities of this vehicle (with reference to conventional airplane design methods). Confidence in the approach taken has been enhanced by using the Boeing weights methodology to calculate the weight of the BAC ALCM"B" - a configuration for which detailed weights data are available.

Design considerations influencing the weight calculation include:

- o Airframe Construction

Wing - The wing is constructed with a center core covered with a 0.2-inch-thick skin of polyimide/graphite material. Forward and aft of the center core are sections of chord 20 inches that have a similar structure but with a thin (0.05-inch) skin of titanium bonded to it. Forward and aft of this region are 14-inch chord sections constructed of a honeycomb core with a 0.08-inch titanium skin. The leading and trailing edges and wing tip are made of solid titanium.

Fuselage - The fuselage is of skin-frame construction composed of polyimide/graphite material with an outer skin of 0.02-inch gauge titanium. Between the polyimide/graphite and titanium skins in a 0.125-inch layer of insulation; this keeps the fuel temperature below 350°F.

- o Radar absorbing material is applied at appropriate parts of the airframe

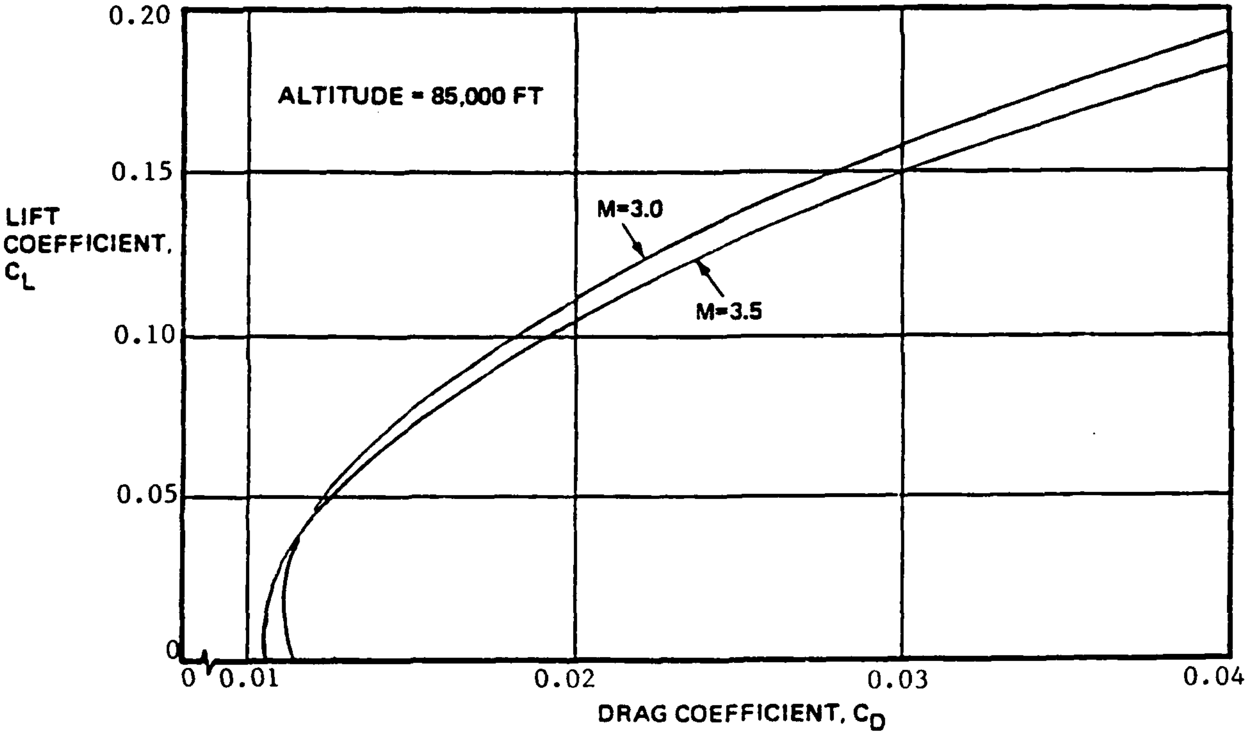


Figure 3-16. Cruise Missile Drag Polars

SICN GROUP WEIGHT STATEMENT WTS 01-SEP-84 VERSION 11-DEC-84	WEIGHT-DES
WING TAIL BODY NACELLE + AIR INDUCTION INSULATION - RAM ATTACH (AIR CARB + BOOST)	944. 119. 1493. 65. 410. 100.
TOTAL STRUCTURE	3132.
ENGINE, THRUST REV + EXHAUST FUEL SYSTEM	765. 392.
TOTAL PROPULSION	1157.
FLIGHT CONTROL HYDRAULIC, PNEUMATIC + ELECTRIC AIR COND + ANTI-ICING LOAD + HANDLING	233. 173. 97. 10.
TOTAL FIXED EQUIPMENT	513.
WEIGHT EMPTY OIL + UNUSABLE FUEL	4801. 42.
NON-EXP USEFUL LOAD	42.
OPERATING WEIGHT PAYLOAD FUEL	4844. 400. 3356.
GROSS WEIGHT	8600.

Figure 3-17. Cruise Missile Weight Statement

- No crew or crew accommodation equipment is included

- No landing gear is present

- High density fuel (JP-10 synthetic, hydrocarbon) is employed (no ullage allowance is included in the fuel tanks; a bellows arrangement allows for fuel expansion)

- Load factor was selected to be 12 to allow for safe carriage by appropriate aircraft.

3.3.4 Propulsion

The supersonic intercontinental cruise missile was designed to use a high temperature, nonaugmented, turbojet engine. The cruise missile has been designed with a Mach 3.5 two-dimensional, mixed compression inlet system. The inlet features an initial compression ramp of 7°. A variable ramp system was used to provide efficient external compression at the design condition and low spillage drag at off-design conditions. Porous boundary layer bleed surfaces were located on all four sides of the internal duct. The bleed was passed into three compartmented bleed plenums and exhausted overboard. A bypass system was also included for engine inlet matching and to enhance inlet restart capability.

A fixed geometry, expansion/deflection nozzle was selected for the cruise missile to reduce the overall engine installation length. The nozzle was designed to use a Prandtl-Meyer expansion that is formed from the nozzle throat to the exit plane about a base plug. The resulting supersonic contour was short, so that frictional losses were lower than for conventional nozzles. This gain was offset, however, by the drag due to low pressure on the base plug.

It is important to note that the cruise missile has been designed for a Mach 3.5, 85,000-ft supersonic cruise condition and that the fixed area expansion/deflection nozzle has been drawn for a minimum power setting. Cruise thrust and SFC are shown in Figures 3-18 and 3-19.

3.3.5 Performance

The portion of the cruise missile mission that employs gas turbine propulsion is shown in Figure 3-20. The gas turbine is started at a Mach number of 3.5 at 85,000 feet altitude.

The mission consists of a cruise-climb to about 98,000 feet at constant Mach number. The mission ends at the point where the

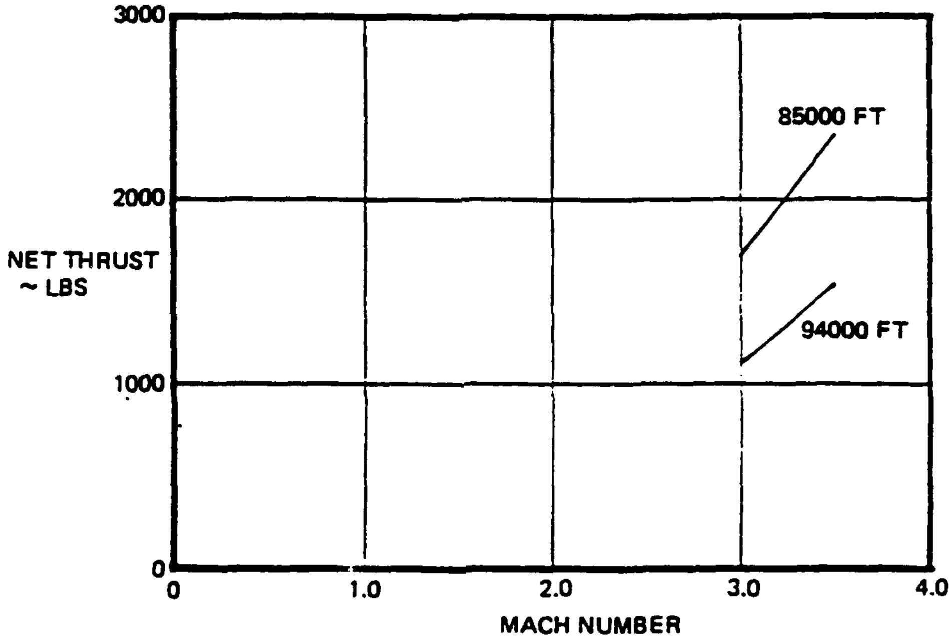


Figure 3-18. Cruise Missile Thrust Available

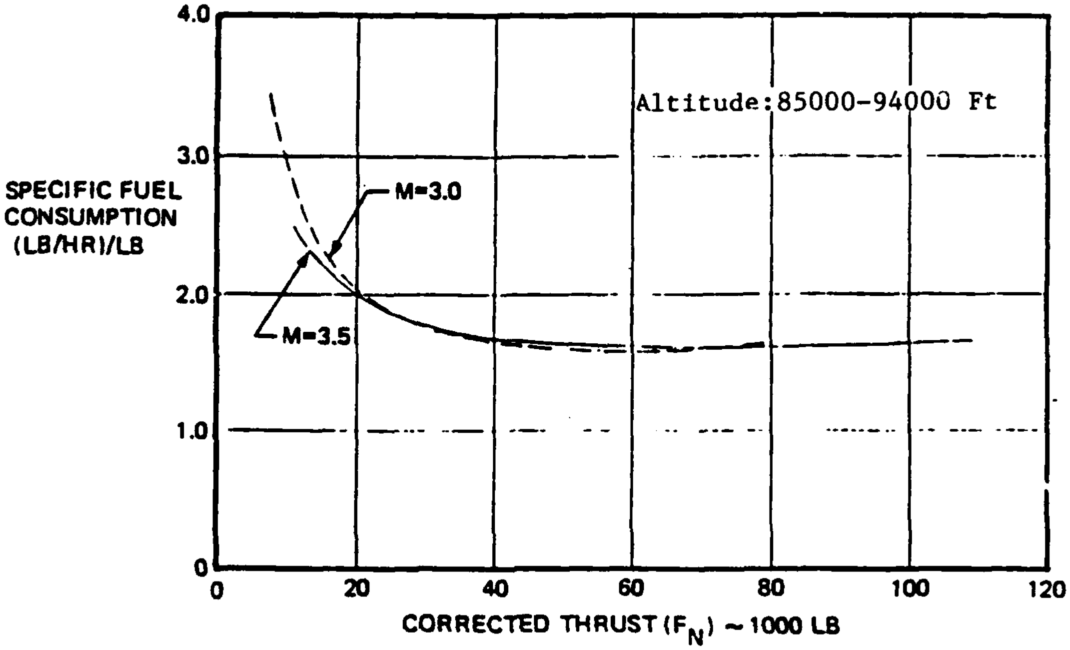


Figure 3-19. Cruise Missile Specific Fuel Consumption

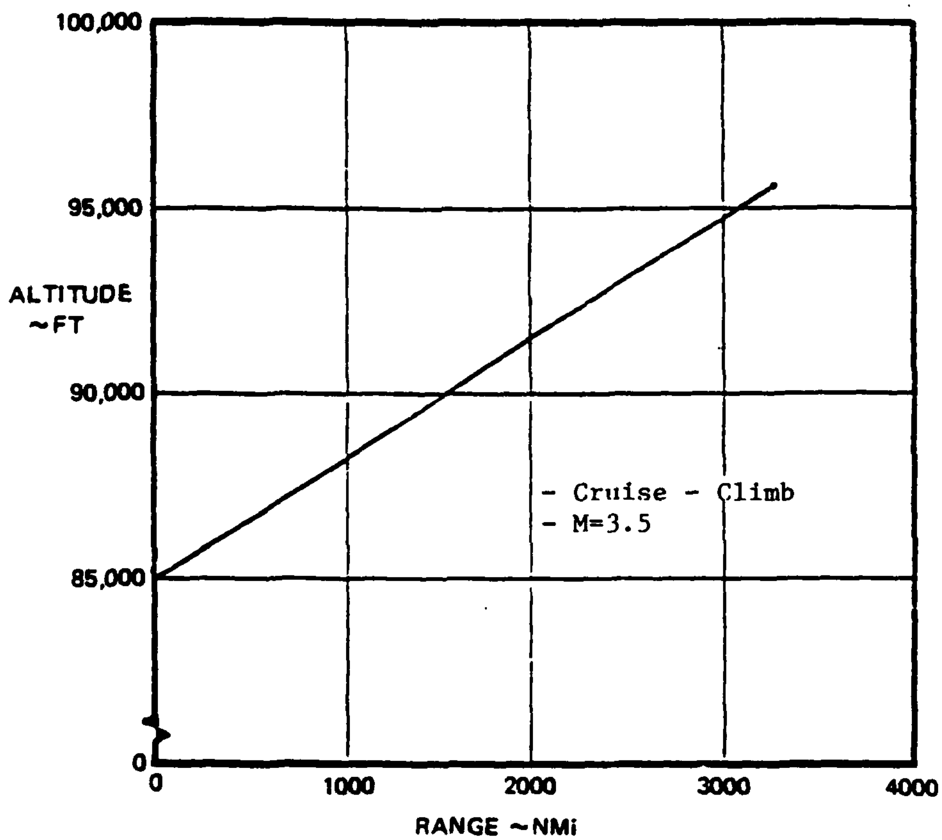


Figure 3-20. Cruise Missile Design Mission

fuel is all expended; at this point, the ballistic payload is released over the target area. The airframe is not recovered.

The cruise is carried out at optimum lift-drag ratio and at a constant power setting (maximum dry thrust). The resulting range factor varies from about 6900 to 7100 nautical miles over the extent of the cruise.

3.4 Long Range Transport - Model 1046-103

3.4.1 Concept Description

This aircraft is shown in the three-view engineering drawing in Figure 3-21.

The aircraft has an overall length of 220.8 feet and wingspan of 206.84 feet. The wing has a leading edge sweepback angle of 37°, a reference area of 4754 square feet, an aspect ratio of 9, and wing thickness-to-chord ratio that varies from 0.12 at the root to 0.08 at the tip. High lift for takeoff and landing is provided by full-span leading edge slot and double-slotted trailing edge flaps.

The body is designed to carry a payload of 200,000 pounds consisting of heavy and/or outsized cargo. The cargo compartment is 142.8 feet long, 17.5 feet wide, and has a maximum height of 13.5 feet. The body has cargo doors and a loading ramp under the upswept rear fuselage and a hinged nose thus providing a drive-through capability. The high-flotation landing gear (with kneeling capability for easy loading) is housed in pods located on the lower part of the fuselage.

The fuselage volume is totally dedicated to cargo so no fuel is carried there. All the fuel is stored in the wing.

The aircraft has conventional, horizontal and vertical tails of area 1060 and 786 square feet, respectively. Elevators and rudder of 30% chord provide flight control surfaces.

Propulsion is provided by four nacelle-housed P&W parametric turbofan engines of bypass ratio 5.74 sized to produce 30,050 pounds of static thrust.

The design constraints imposed on this configuration include:

o Payload	200,000 pounds
o Range	4,600 nautical miles

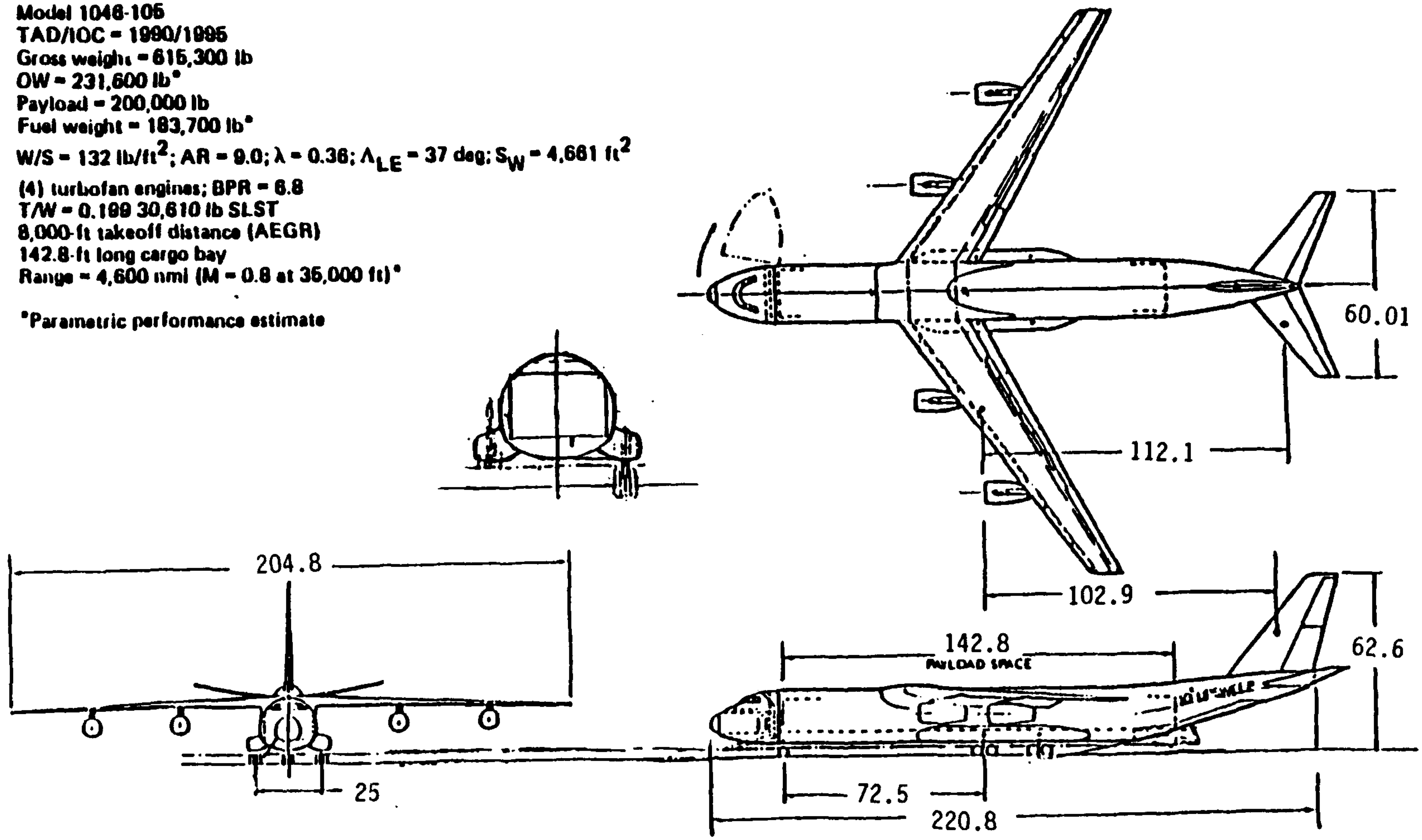


Figure 3-21. Long Range Military Logistics Transport

- o Critical Field Length — 8,000 feet
- o Load Factor — 2.5
- o R/C with OEI — 100 feet/minute
- o No fuel stored in fuselage

3.4.2 Aerodynamics

The aerodynamic characteristics of the Model 1046-103 in the form of drag polars for important Mach numbers are shown in Figure 3-22.

3.4.3 Weights

Figure 3-23 shows the weight statement of the Model 1046-105. Weight estimation is consistent with a TAD of 1990.

3.4.4 Propulsion System

The nonaugmented turbofan engine selected for the Model 1046-103 transport concept was chosen engines investigated in the Advanced Technology Engine Studies (ATES) program. The engine cycle included a bypass ratio of 5.74, an overall pressure ratio of 35.0, and a combustor exit temperature of 2600°F.

A turboprop engine, the Pratt & Whitney STS679, has also been supplied for use with the Model 1046-105. This three-spool advanced technology engine features a two-axial stage, one centrifugal stage, high compressor driven by a single-axial stage turbine, a four-axial stage low compressor driven by a single-stage turbine, and a gearbox driven by a three-stage free turbine. The overall pressure ratio of the engine is 27.5, the combustor exit temperature was 2379°F, and the speed of the power turbine is 10,960 RPM.

The propeller selected for use with the STS679 was chosen from a previous Boeing in-house study of near-term and advanced propellers supplied by Hamilton Standard. Propeller tip speed and loading were also selected based on this study. The system chosen was a counter rotating prop fan. This small diameter, highly loaded, multibladed, variable pitch, unducted fan has been designed for use on aircraft with cruise speeds up to Mach 0.85.

Thrust and SFC of the installed turbofan engine are shown in Figures 3-24 and 3-25, respectively.

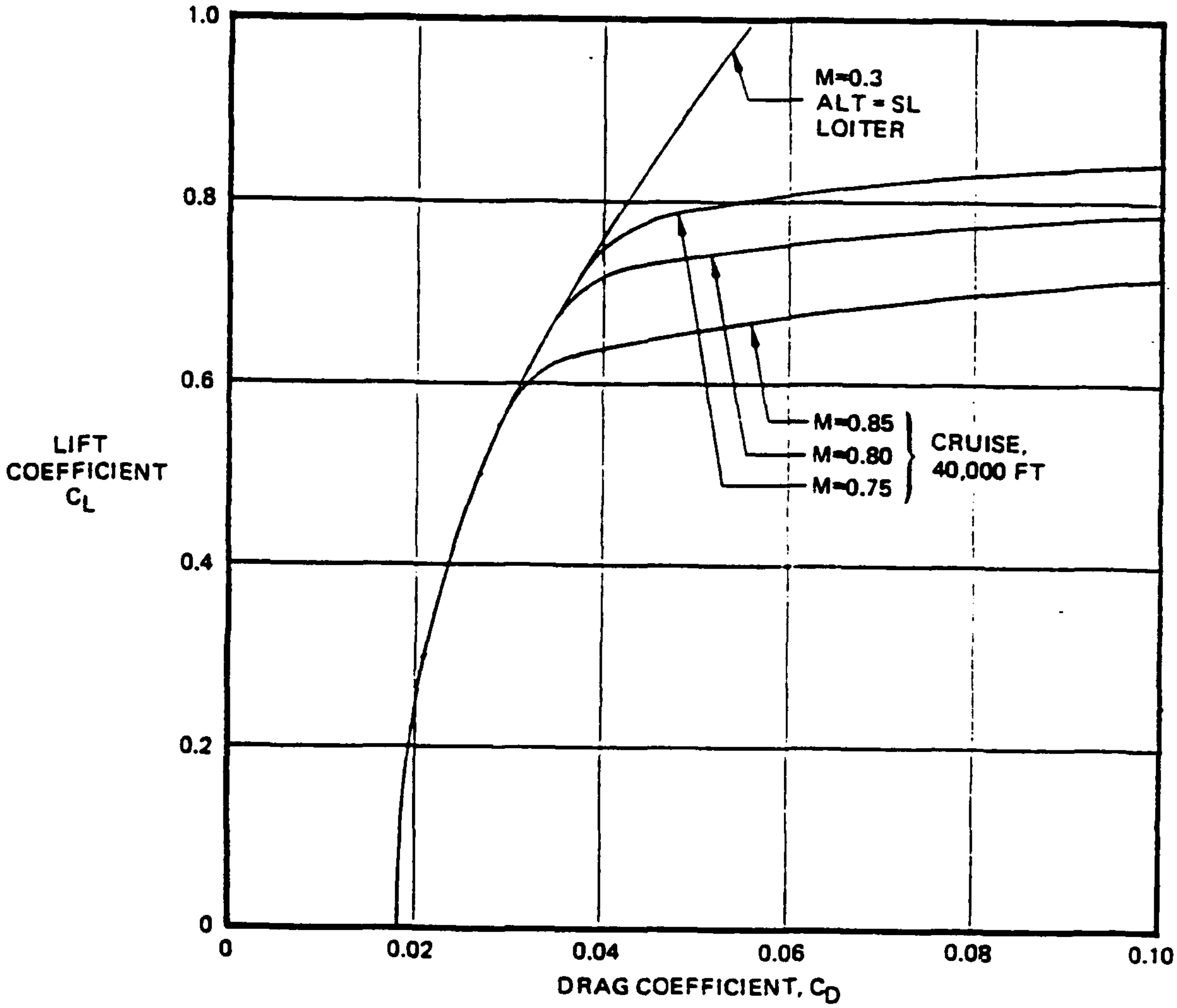


Figure 3-22. Long Range Transport Drag Polars

Group Weight Statement IADSWT2 07/01/77 Version Aug 27	Weight Lbs	Design Data	
Wing	60321	Gross Wt	623360
Horizontal Tail	5294	Design Wt	623360
Vertical Tail	4128	Landing Wt	528000
Body	70478	Load Factor	3.75
Main Gear	29716	Mach SL	.48
Nose Gear	3219	Mach Max	.00
Auxiliary Gear	0	Max O	0
Nacelle or Eng Section	7511	VStall	107
Air Induction	0	CLMAX	2.84
		Wing	
Total Structure	175668	SGross	4758
		SEXP	3840
Engine + Accessories	22507	Aspect Ratio	9.00
Thrust Reversers	3117	Taper Ratio	.36
Exhaust + Deflectors	0	TOC Root	.12
Fuel System	2317	TOC Tip	.08
Engine Control	100	Sweep E.A.	33
Starting System	400	H Tail	
		SGross	1061
Total Propulsion	28440	SEXP	921
		Aspect Ratio	4.50
Flight Control	7423	Taper Ratio	.36
Auxiliary Power Plant	931	TOC Root	.09
Instruments	860	TOC Tip	.09
Hydraulic + Pneumatic	2139	Sweep E.A.	29
Electrical	3528	Tail Arm	112
Avionics	3451	V Tail	
Armament	0	SGross	736
Furnishings + Equip	4483	Aspect Ratio	1.65
Air Cond + Anti-Icing	3103	Taper Ratio	.36
Photographic	0	TOC Root	.10
Load + Handling	0	TOC Tip	.10
		Sweep E.A.	25
Total Fixed Equipment	25917	Tail Arm	102
		Body	
Weight Empty	230025	Swet	12580
		Length	221
Crew	645	Width	21.70
Unusable Fuel	389	Depth	19.60
Oil + Trapped Oil	441	Delta P	8.58
Tare Weight	0	Landing Gear	
Operating Items	0	NG Length	90
Crew Equipment	90	MG Length	130
		MG Tires	16
Non-Exp Useful Load	1565	Propulsion	
		SLST	30084
Operating Weight	231590	SFC	.58
		Tank Volume	28970
Payload	200000	Systems	
Passengers + Baggage	0	KVA Reqd	202
Fuel	191770	Volume Pres	50690
Gross Weight	623360		

Figure 3-23. Long Range Transport Weight Statement

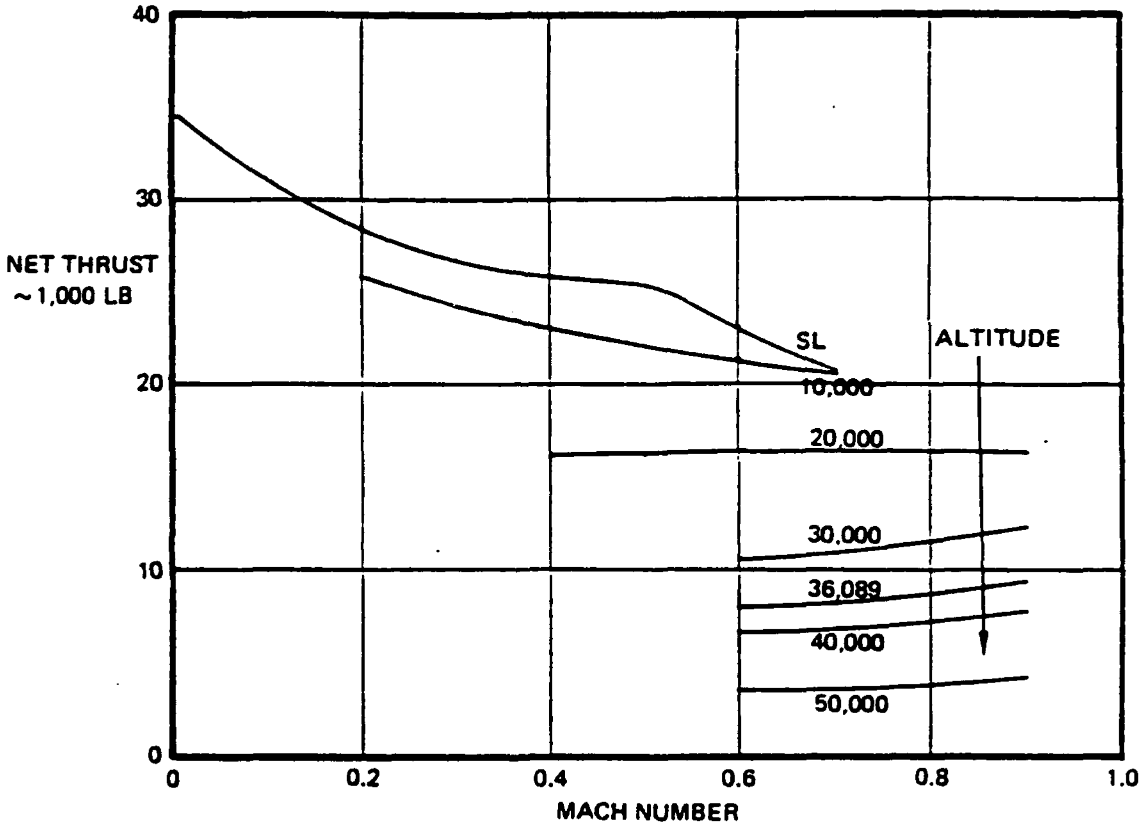


Figure 3-24. Long Range Transport, Cruise Thrust

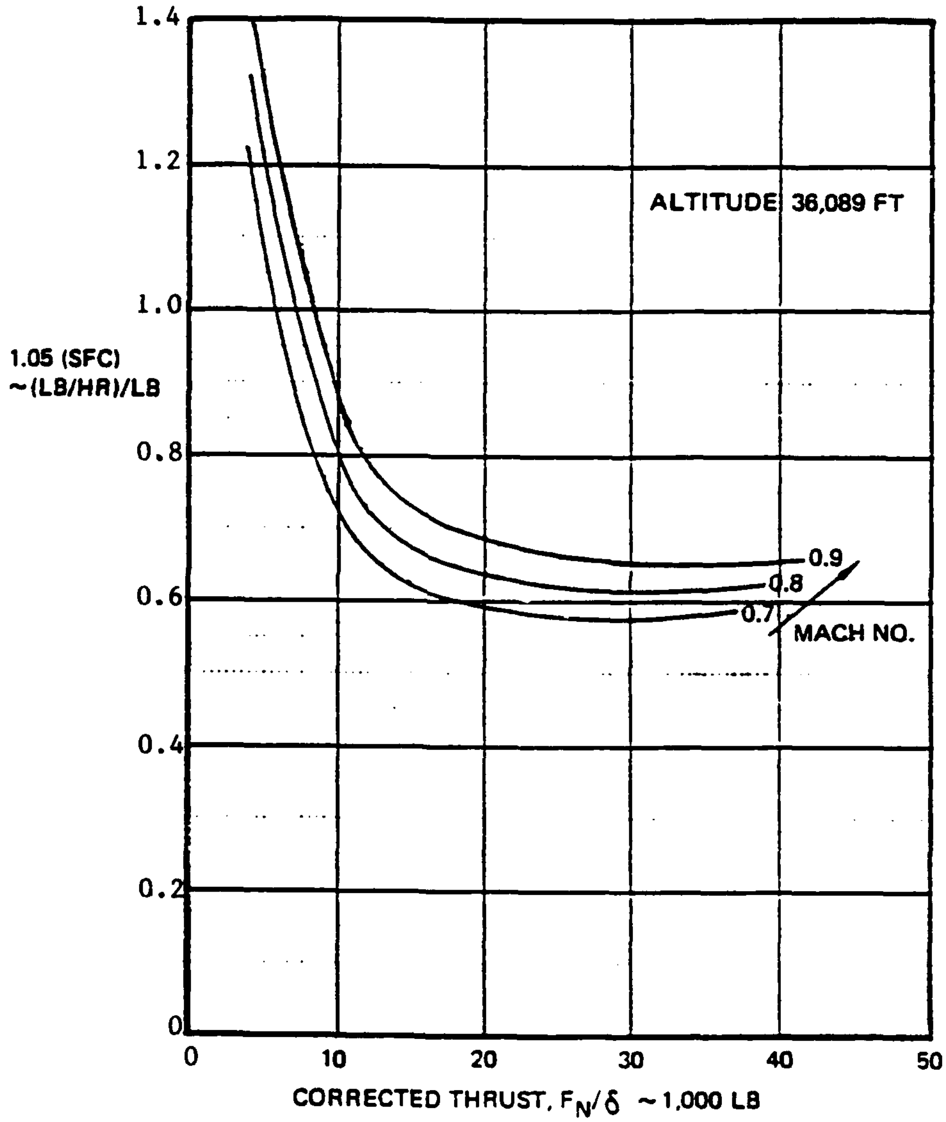


Figure 3-25. Long Range Transport, Cruise SFC

3.4.5 Performance

The Model 1046-103 was designed to carry a payload of 200,000 pounds of cargo over a range of 4600 nautical miles. To minimize aircraft size and cost the mission is flown at optimum altitude and cruise Mach number. The mission is illustrated in Figure 3-26.

A summary of the designed mission is shown in Figure 3-27.

3.5 Lightweight Fighter - Model 985-213 (Modified)

3.5.1 Concept Description

The vehicle consists of a blended wing-body configuration with twin vertical tails mounted on the wings at about the 3/4 span location (Figure 3-28).

The overall length of the aircraft is 44 feet and the wingspan is 19.7 feet. The wing has a NASA SCAT 15 plan form with 74° leading edge sweep, with an aspect ratio of 1.46, taper of 0.19 and a reference area of 266 square feet. The wing thickness varies from 4% at the root to 3% at the tip. Wing camber is variable throughout the flight envelope.

The aircraft carries a one-person crew in a low-profile cockpit at the design takeoff gross weight of 12,500 pounds, the design wing loading is 47 pounds per square foot, and the thrust/weight ratio if 1.32.

Wing structure is skin and multispar construction of graphite composite material. The structure is designed for a load factor of 7.33 g's at the flight weight of 12,500 pounds, a dynamic pressure placard of 2133 pounds per square foot (Mach 1.2 at sea level) and a Mach 2.2 dash capability at altitude.

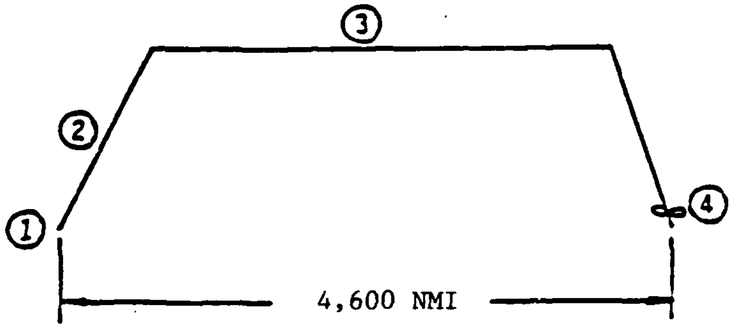
Air-to-air weapon capability consists of two lightweight (CLAW) missiles mounted semisubmerged on the upper aft fuselage. A 20-mm gun and 250 rounds of ammunition are carried internally.

3.5.2 Aerodynamic

Estimated aerodynamic characteristics of the unmodified Model 985-213 are presented in Figure 3-29 through 3-31. Figure 3-29 shows the detailed breakdown of drag-at-zero-lift for three flight conditions.

Trimmed drag polars for typical subsonic and supersonic flight conditions are shown in Figures 3-30 and 3-31,

AIRLIFT COMBAT MISSION



1. WARMUP/TAKEOFF - 5 MIN.
2. CLIMB
3. CRUISE - OPT MACH/ALTITUDE
4. LOITER - 30 MIN.

Figure 3-26. Long Range Transport Design Mission Profile

AIRLIFT MISSION SUMMARY

	WT ~LB	MACH	ALT ~FT	DIST or TIME	FUEL ~LB	L/D	CL	P.S.	SFC	ROC ~FT/M
TAKEOFF	623360		0		4184			2.000	.350	
CLB-EAS	619176	.497	0	50	5925	16.792	.354	1.000	.570	2384
MACH-CLB	613251	.671	15649	245	13942	16.765	.351	1.000	.000	1996
CRUISE	599308	.833	33008	0	0	18.824	.474	.897	.634	300
CRUISE	518504	.833	35787	4453	161610	18.572	.466	.896	.627	305
LOITER	437699	.272	0	.500	6111	18.645	.842	.250	.524	

Figure 3-27. Long Range Transport Design Mission Summary

GEOMETRY	WING	VERT
• WING AREA	266	38
• L.E. SWEEP	74	65
• ASPECT RATIO	1.46	0.68
• TAPER RATIO	0.19	0.32
• THICKNESS RATIO	0.04→0.03	0.046→0.03
• MAC, INCHES	227	—

POINT DESIGN WEIGHTS	
• DESIGN MISSION	12,500 POUNDS
• OVERLOAD MISSION	16,780 POUNDS
• OPERATING WEIGHT	8,530
• FULL INTERNAL FUEL	3,630
• WEIGHT EMPTY	7,910
• PAYLOAD – DESIGN MISSION	340
• PAYLOAD – A/G MISSION	4,000
• AVIONICS – DAY A/A	390

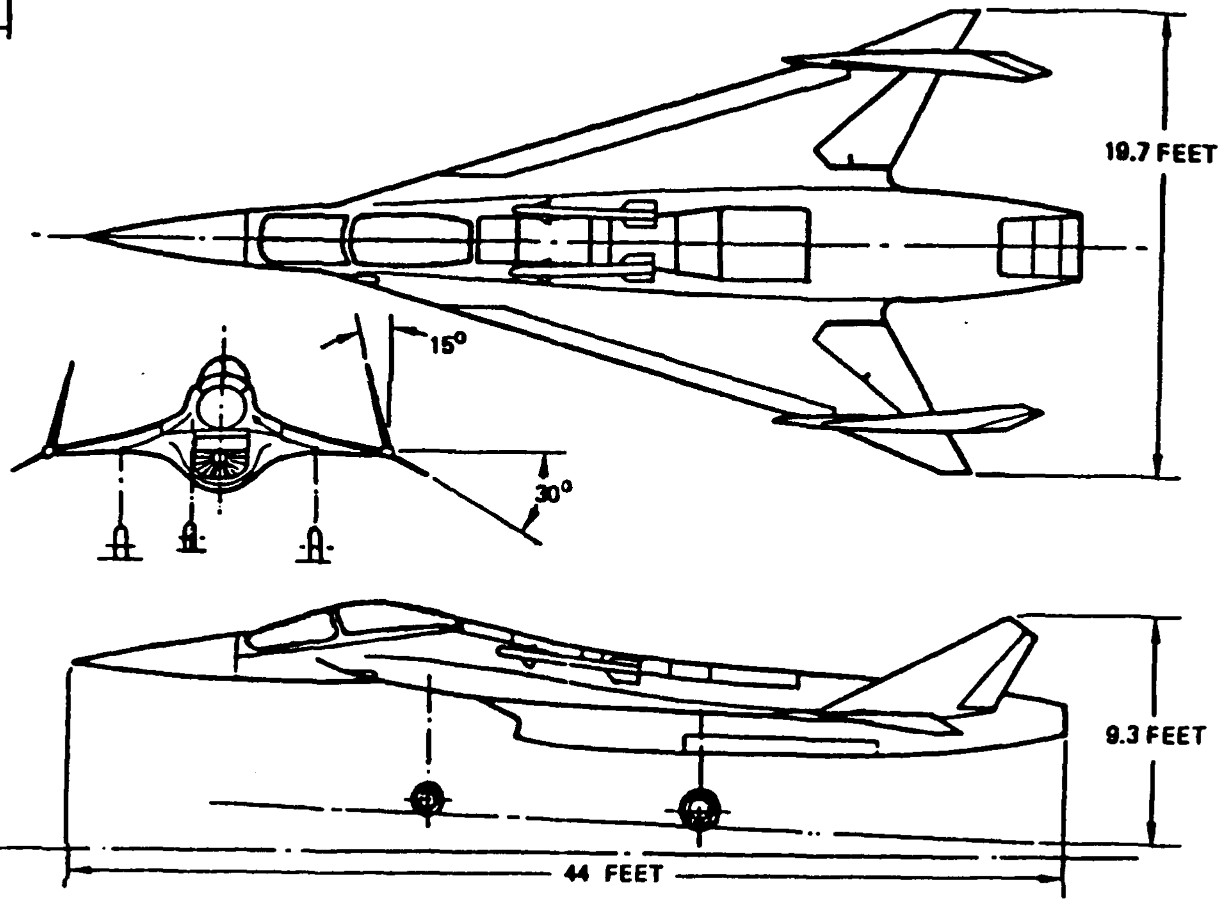


Figure 3-28. Lightweight Fighter, Model 985-213

Components	M=0.9 30,000 Feet	M=1.2 30,000 Feet	M=1.8 50,000 Feet
Wing (A_{wet}=383 ft^2)	0.00354	*	0.00518
Skin friction	0.00349	0.00319	0.00305
Form	0.00005	----	----
Wave	-----	*	0.00213
Body (A_{wet}=324 ft^2)	0.00340	*	0.00367
Skin friction	0.00247	0.00226	0.00214
Form	0.00009	----	----
Wave	----	*	0.00182
Interference (wing-body)	0.00084	*	-0.00029
Vertical tails (A_{wet} = 74 ft^2)	0.00186	*	0.00099
Skin friction	0.00077	0.00071	0.00068
Form	0.00001	----	----
Wave	----	*	0.00035
Interference (vertical-wing)	0.00108	*	-0.0004
Excrescence	0.00150	0.00220	0.00183
Inlet diverter**	0.00070	0.00110	0.00090
Misc Items	0.00133	0.00333	0.00333
Canopy	0.00025		
Gun fairing	0.00010		
UHF/IFF antennas (2)	0.00005	2.5 Factor applied to M = 0.9 estimate	2.5 Factor applied to M = 0.9 estimate
Fuel tank vents (4)	0.00001		
Nav Beacon	0.00001		
Air data probe	0.00011		
Missiles (2 semi-submerged)	0.00080		
Total non-lifting drag	0.01333	0.01875	0.01690
Camber and trim drag at C_L = 0	0	0.00770	0.00780
Total drag at C_L = O, C_{DO}	0.01333	0.02645	0.02470

S_{ref} = 260 feet^2
* Not itemized; total C_{DW} @ M = 1.2 = 0.00412

Figure 3-29. Zero Lift Drag Summary

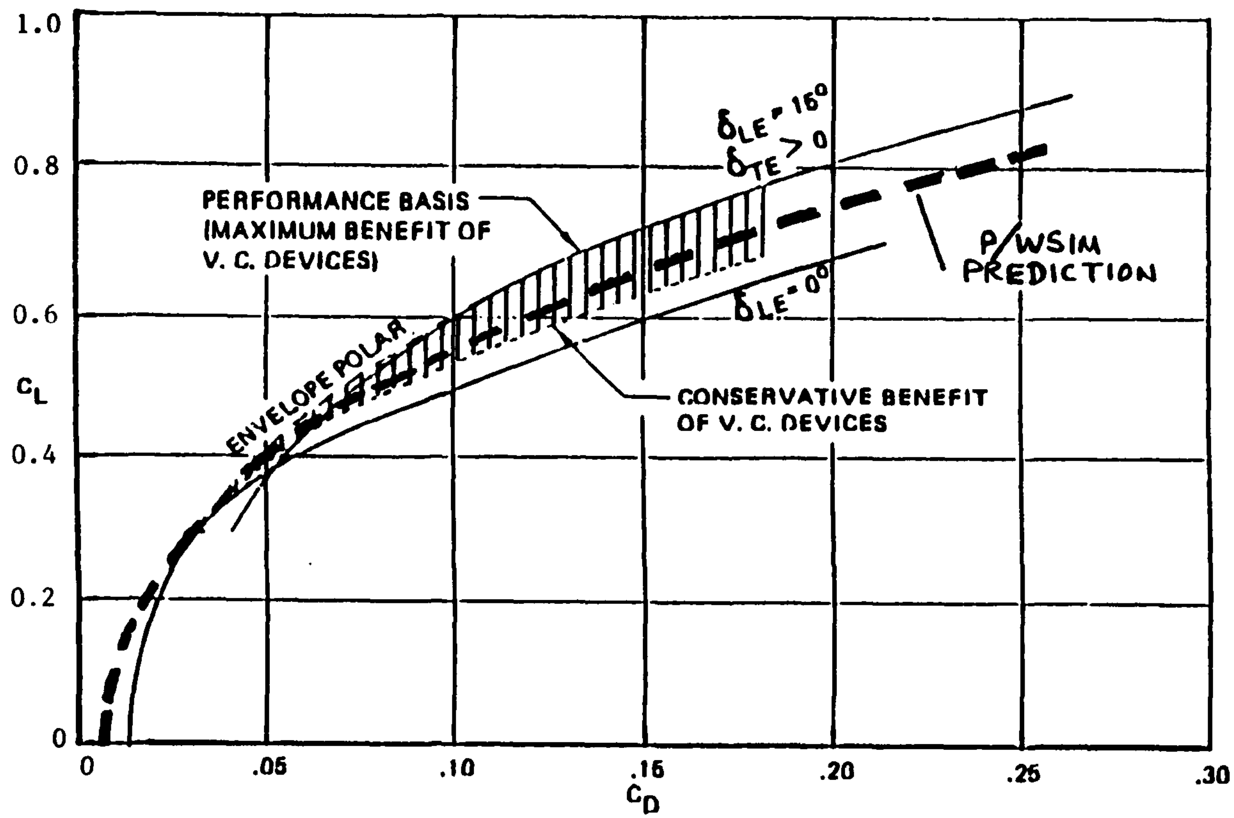


Figure 3-30. LES-213 Subsonic Trimmed Drag Polar

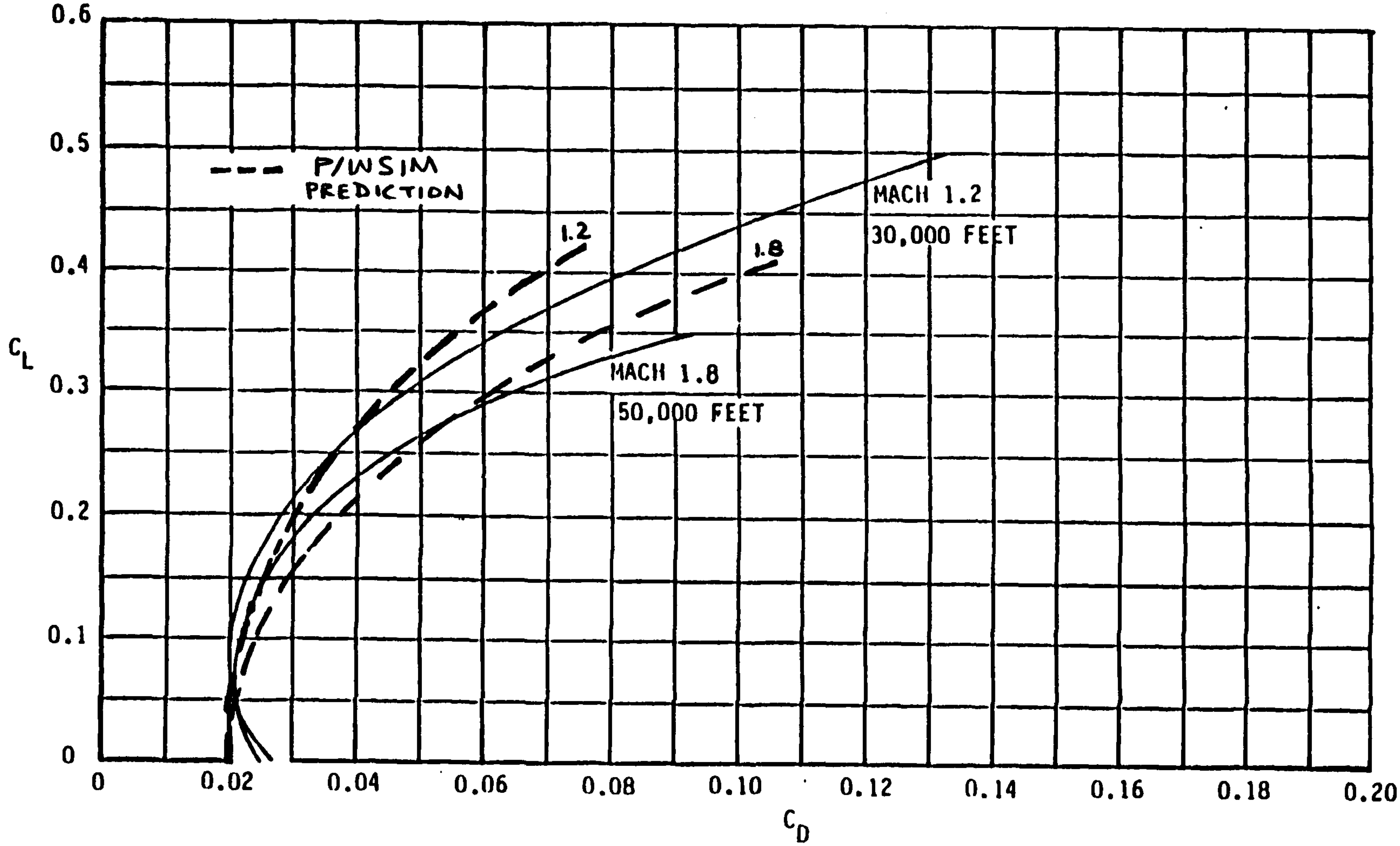


Figure 3-31. LES-213 Supersonic Trimmed Drag Polars

respectively. The drag of the new aircraft should be close to that of the original; differences due to the aft-body reference drag changes are to be expected because of the different installation.

3.5.3 Weights

The weight statement for the 985-213 is shown in Figure 3-32. Weight estimation ground rules and assumptions are listed below:

- o The majority of aircraft structure is advanced composites (graphite-epoxy)
- o Airframe Integrated Nozzle
- o Fly-by-wire surface controls
- o Avionics equipment in compliance with statement-of-work requirements
- o Semisubmerged CLAW missiles (2)
- o Final aircraft geometry is the result of aerodynamic and weight parametric trade studies and represents the best compromise for overall performance
- o Lightweight M-197 20-mm Gatling gun with gas drive
- o Judicious location of gun, ammunition, missiles, and fuel such as to minimize CG gravel as these items are expended
- o Fuel pumping for trim control.

3.5.4 Propulsion

Uninstalled engine performance was computed using the Pratt & Whitney Aircraft parametric engine cycle deck, CCD 1178-08.00. The engine is a minimum bypass ratio, dry turbofan having a max dry uninstalled thrust of 16,500 lb sea level static. The engine cycle characteristics are bypass ratio (BPR) = 0.2, overall pressure ratio (OPR) = 26, turbine inlet temperature (TIT) = 3000°F.

The inlet is located under the fuselage, centerline mounted. it is a two-dimensional, external compression inlet utilizing a variable ramp, four-shock system.

This inlet has two movable external ramps, a 7.30 initial ramp angle, a boundary layer control bleed system consisting of

	WEIGHT LBS.
WING	1180
HORIZONTAL TAIL	
VERTICAL TAIL	100
BODY	1520
MAIN GEAR	380
NOSE GEAR	110
LAUNCH AND RECOVERY GEAR	
ENG SECTION OR NACELLE	360
STRUCTURE	(3650)
ENGINE AND EXHAUST	2200
THRUST REVERSER	
ENGINE ACCESSORIES	50
ENGINE CONTROLS	80
STARTING SYSTEM	100
FUEL SYSTEM	340
PROPULSION	(2770)
FLIGHT CONTROLS	260
AUXILIARY POWER PLANT	
INSTRUMENTS	70
HYDRAULIC & PNEUMATIC	120
ELECTRICAL	270
AVIONICS	390
ARMAMENT	40
FURNISHINGS & EQUIPMENT	180
AIR CONDITIONING	120
ANTI-ICING	10
LOAD & HANDLING	30
FIXED EQUIPMENT	(1490)
WEIGHT EMPTY	7910
CREW	200
UNUSABLE FUEL	30
OIL AND TRAPPED OIL	60
EXTERNAL TANKS	
GUN INSTALLATIONS	260
WEAPON INSTALLATIONS	60
CREW EQUIPMENT	10
NON-EXP USEFUL LOAD	(620)
OPERATING WEIGHT	8,530
FUEL - INTERNAL	3,630
FUEL - EXTERNAL	
AMMUNITION-250 RNDS 20MM	180
CLAW MISSILES	160
GROSS WEIGHT (MISSION T.O)	12,500
BASIC MISS FLT DES WT	10,400
FULL INTERNAL FUEL	3,630

Figure 3-32. Weight Statement

porous bleed on the second and third ramp surfaces, sideplates, and throat bleed slot located aft of the normal shock. The throat slot also acts as a bypass to remove excess inlet airflow for matching engine airflow demand with inlet supply. The inlet capture area is 4.44 ft^2.

An engine mounted 2-D/C-D nozzle which incorporates a fixed throat and a variable exit area was utilized for efficient engine operation.

3.5.5 Performance

The aircraft was configured to provide low drag at the design Mach number of 1.8. A design mission was specified (see Figure 3-33) that involved flight at altitudes limited to 50,000 feet (a pressure suit limit). Mission characteristics are summarized in Figure 3-34.

3.6 Carrier Air Vehicle/Transatmospheric Vehicle

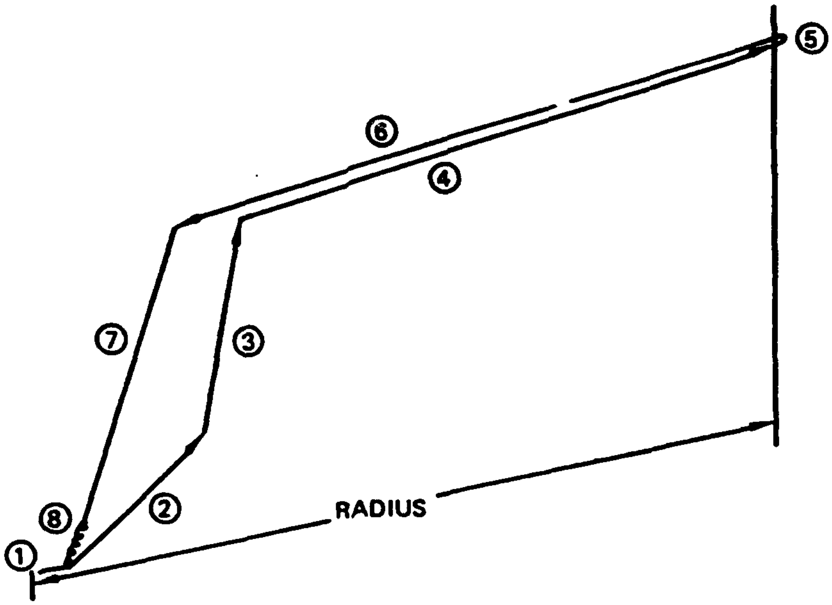
3.6.1 Concept Description

The Model 896-111 is a two-stage-to-orbit system with both stages being recoverable (Figure 3-35).

The orbiting vehicle is carried in a cavity in the underside of the fuselage of the first stage. This concept minimizes the requirement for the large amount of ground-support equipment normally associated with today's conventional vertical takeoff rocket launch system. The proposed system utilizes a horizontal takeoff and landing mode.

For mating, the booster and orbiter are each towed to an "alert pad" and the vehicles aligned with their longitudinal centerlines coincident with each other. The orbiter is then towed forward into the booster body cavity and mechanically joined to the booster. The orbiter landing gears are retracted, and the booster orbiter combination is towed to the LOX/LH2 servicing facility which is adjacent to the TAV pad to allow all cryogenic loading and replenishment to be controlled in one area. After completion of the takeoff, climb, and separation, the booster would return to the base to be recycled for any necessary maintenance.

The CAV is illustrated in Figure 3-36. The two-man crew and aircraft subsystems are located in the forward body. The two cylindrical LH2 fuel tanks are paired in the forward fuselage with the LOX tank pair located directly to the rear. The nose landing gear is located forward and below the LH2 tankage. The



① TAKEOFF FUEL ALLOWANCE

- 2.5 MIN IDLE FUEL FLOW RATE
- ½ MIN MAX POWER FUEL FLOW RATE
- MAX POWER ACCEL TO CLIMB SPEED

② MAXIMUM POWER CLIMB (q = 2,132 psf)

③ MAXIMUM POWER CLIMB (M=1.8)

④ SUPERSONIC CRUISE (M=1.8, h=50,000 FT)

⑤ COMBAT – 1 FULL POWER TURN

⑥ SUPERSONIC CRUISE (M=1.8, h=50,000 FT)

⑦ MINIMUM POWER DESCENT

⑧ RESERVES; 20 MIN SEA LEVEL LOITER OPTIMUM MACH

Figure 3-33. *Design Mission Profile*

	RADIUS = 200 NMI		
	INITIAL WEIGHT - lB	DISTANCE NMI	FUEL LB
TAXI	12,500	0	160
TAKEOFF	12,340	0	50
ACCELERATE	12,290	2	60
CLIMB	12,230	31	660
CRUISE	11,570	167	690
COMBAT	10,880	0	470
EXPEND PAYLOAD	10,410	0	---
TURN AROUND	10,070	5	220
CRUISE	9,850	195	810
LOITER	9,040	0	510
OW	8,530		(3630)

Figure 3-34. Mission Summary

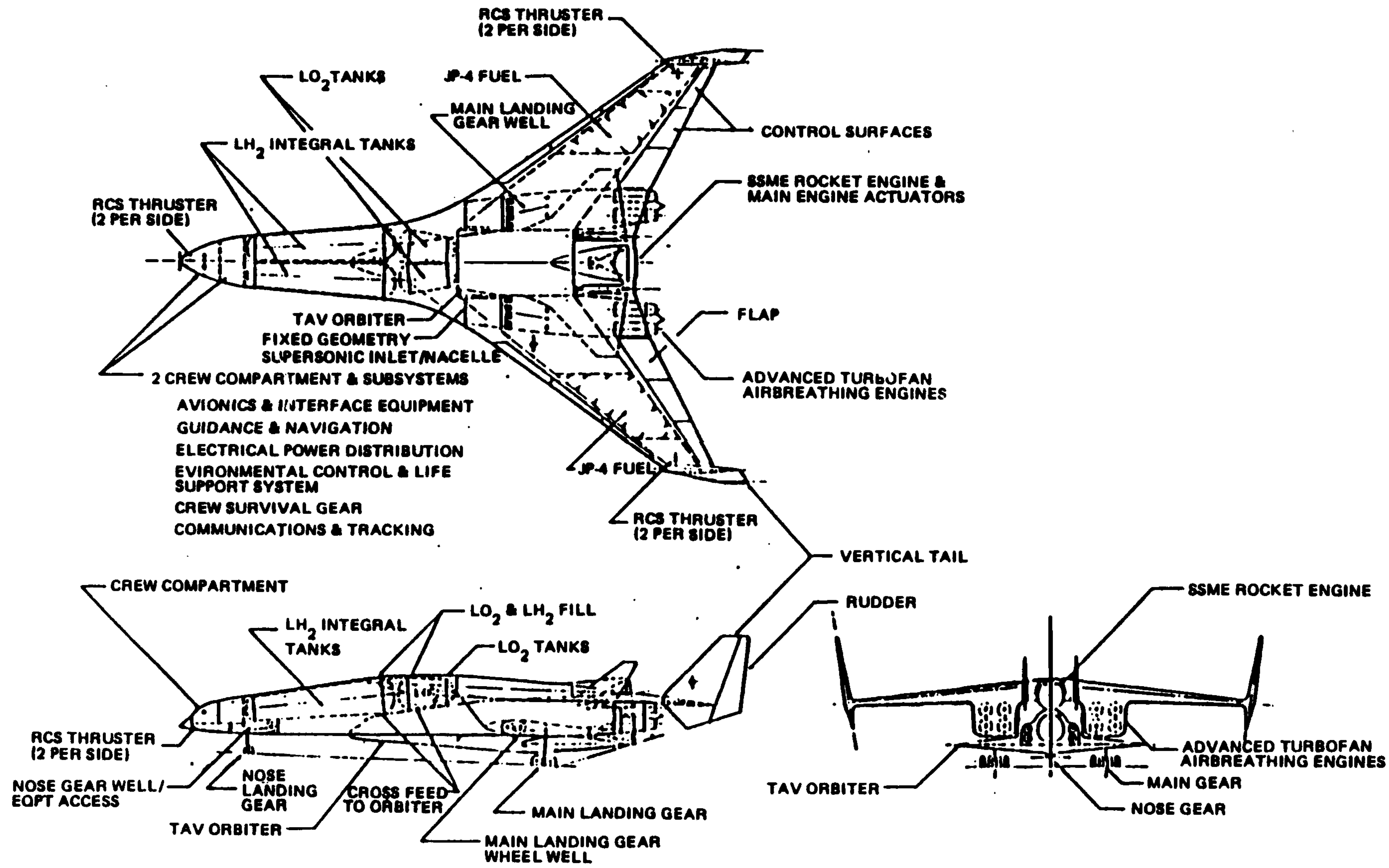


Figure 3-35. CAV/TAV System, Boeing Model 896-111

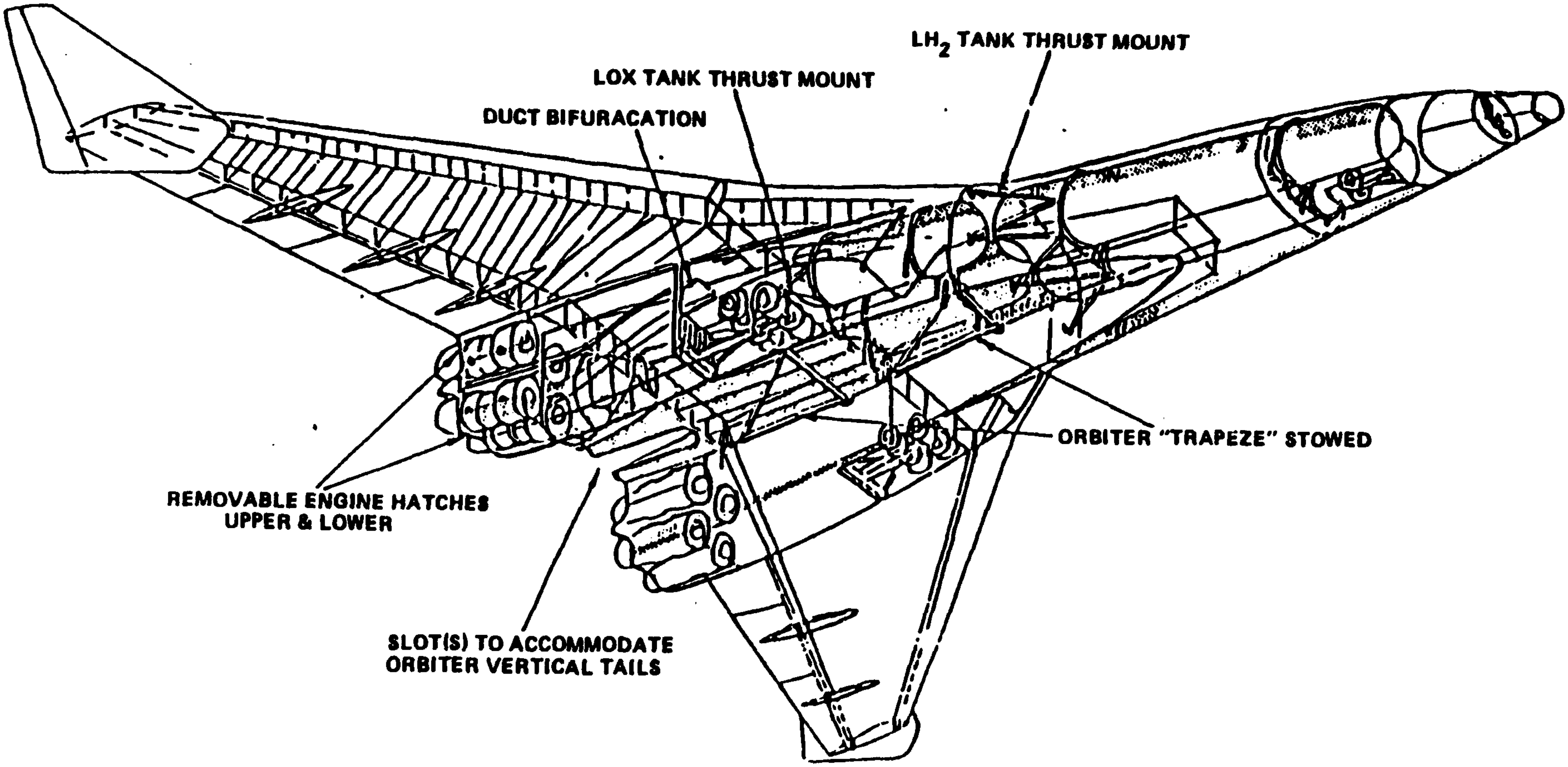


Figure 3-36. Boeing Model 896-111 – General Arrangement

nacelles, with fixed supersonic inlets, are located outboard of the body cavity which accommodates the orbiter. The eight wheeled main landing gear is integral with the nacelle and retracts forward into the lower nacelle when stowed. The wings are mounted high on the fuselage to provide clearance with the underslung orbiter. Wing tip mounted verticals are used to provide directional stability. A single SSME rocket engine is used during boost phase and is located on aircraft centerline at the wing trailing edge.

The booster forward body contains LOX/LH2 rocket propellants and propellant crossfeed system to the orbiter to ensure that the orbiter vehicle propellant tanks are completely filled at stage separation. The JP-4 airbreathing fuel is contained in the outboard wings to reduce the total wing bending moments at the side of body. The booster is designed for a two-man crew. Located forward, aft, and below the crew compartment are the avionics/electronics equipment compartments. ECS equipment, oxygen, and electrical/hydraulic subsystem equipment are located in the fuselage aft of the pilot.

The first stage (booster) utilizes present day state-of-the-art construction.

BODY

Body structure is semimonocoque with frame supported graphite/polyimide honeycomb sandwich skin panels. Two deep aluminum honeycomb beams form the sidewalls of the orbiter recess, carrying twin lower-body longerons and providing vertical shear capability. Attached to the wing by the wing-to-body longeron, these beams extend aft of the wing and form the inboard structure of the airbreathing engines mounting structure. Within the body cavity, the beams carry the pair of trapezes which control the relative movement of the booster and orbiter to ensure clean separation.

The other engine supports are provided by vertical beams attached below the wing, the center one acting as a duct splitter over its forward portion, the outboard one forming the nacelle wall. Further structure is provided by the horizontal duct splitter, which continues aft as a firewall separating the upper and lower engine pairs, providing lateral shear stiffness. Engine removal is effected through individual hatches on the top and bottom surfaces of the nacelle. Removal of any or all of the hatches does not affect the structural integrity of the engine support structure.

The cylindrical LH2 tanks are paired in the forward fuselage, and are link-supported inside the body monocoque. Fore and aft loads are taken by a thrust structure joining the aft tank ring to a body bulkhead which serves to separate the fuel and oxidizer bays, and also forms a manufacturing joint. Aft of this is the LOX tank pair, also link supported, with a thrust structure to the front spar of the wing. Forward of the LH2 tanks are the nose landing gear bulkhead, the equipment and ECS bays, and the crew compartment and capsule.

WING

The high-mounted wing carries the four orbiter attachment points, two each on the front and rear spar center sections. The four-spar wing has graphite/polyimide honeycomb sandwich skins with integral spar caps. Stringers, spar webs, and ribs are graphite/epoxy co-cured. The wing leading edge is a built-up titanium structure with provisions for thermal stress relief. Control surfaces are of graphite epoxy honeycomb.

VERTICAL TAILS

The vertical tails are of similar construction to the wing. The possibility of using split rudders is being studied. This will enhance directional stability in slip-flow conditions by forming wedge-type vertical tail surfaces.

LANDING GEAR

The main landing gear comprises two struts, each carrying an eight wheeled truck, retracting forward into the nacelle lower surface. Vertical loads are reacted to the wing structure by a bulkhead spanning between the inboard beams and the outboard nacelle wall.

The nose landing gear is mounted on the bulkhead ahead of the LH2 tanks, and retracts rearward to lie below the tanks. Provision is made for emergency extension should the hydraulic system fail. Because of the wide spread between takeoff and landing weights, an Adaptive two-stage oleo design is proposed for all three elements of the tricycle landing gear.

3.6.2 Aerodynamics

The drag data shown in Figure 3-37 and 3-38 have been evaluated using several Boeing programs to calculate drag from various sources (skin friction, wave drag, etc.). The data base estimation uses simplified methods that have been calibrated to match the results from the detailed drag analysis.

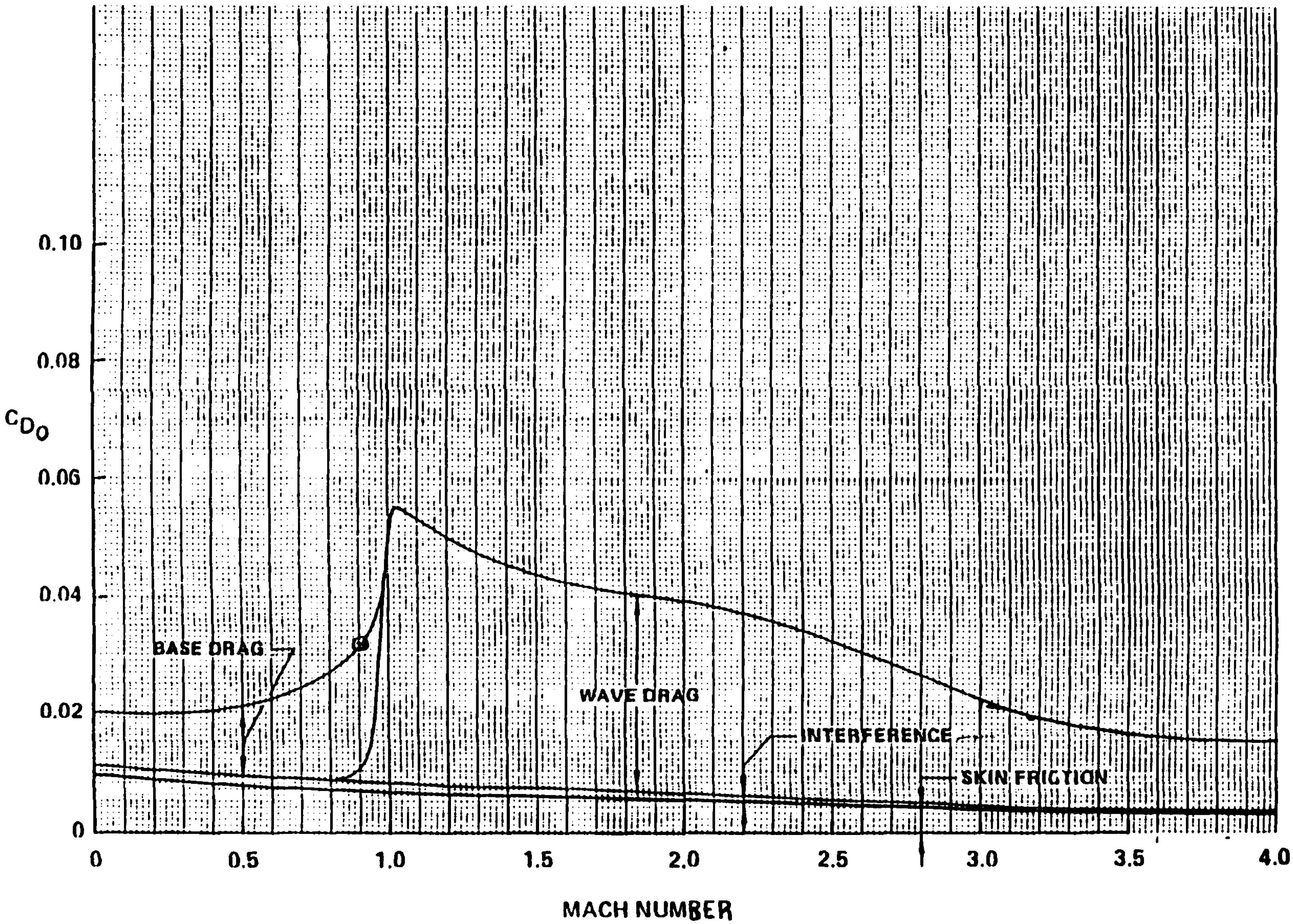


Figure 3-37. Model 896-111 – Drag at Zero Lift

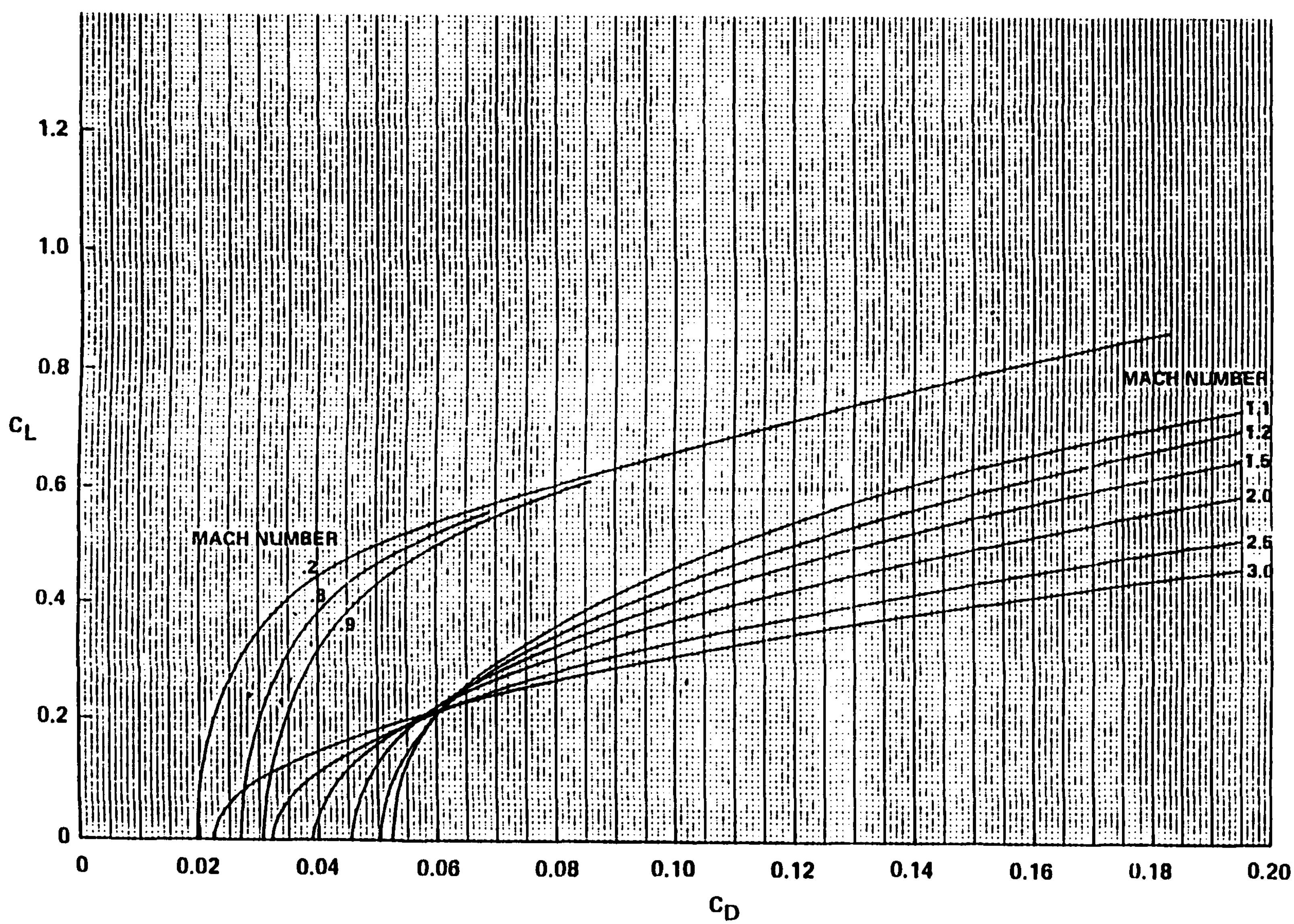


Figure 3-38. Model 896-111 – Drag Polars

The data base program evaluates tables of:

a. drag coefficient at zero lift as a function of altitude and Mach number

b. drag coefficient due to lift as a function of lift coefficient and Mach number

c. 'drag-area' or D/q increments as a function of Mach number

The latter tables allow the mission analysis program to take account of the drag changes that result when:

a. the rocket engines are fired (drag change due to reduced base area)

and

b. the TAV is not attached to the CAV (drag change is due to modified base area and wetted area).

3.6.3 Weights

The weight of the Model 896-111 was estimated using the Boeing Level-1 weight estimating program, PDWTS, for the conventional airplane components; rocket engine, cryogenic systems, etc. were evaluated using detailed analysis of the systems.

A typical weight statement is shown in Figure 3-39.

3.6.4 Propulsion System

The first-stage booster is powered by eight advanced augmented airbreathing engines (F-101 uprated) each producing 35,000 lb static sea level thrust and one SSME rocket engine ($A^*/Ae = 150$) having a vacuum thrust rating of 530,200 lb and an ISPVAC = 463.5 sec using LOX/LH2 propellants. The booster launch system utilizes airbreathing propulsion during the takeoff and climb to 30,000 ft and M = 0.86. At this time, the rocket engines on both stages ignite and operate until reaching 117,500 ft altitude and 3000 ft/s velocity where stage separation occurs.

The airbreathing propulsion system performance in the mission analysis program is calculated from tables of installed thrust, fuel flow, and corrected airflow of the engines. The installed performance data are calculated by the program using

Model 896-111 GROUP WEIGHT STATEMENT WTS 01-SEP-84 VERSION 11-FEB-86	WEIGHT-LBS
Wing	94723.
Tail	9227.
Body	34340.
Alighting Gear	32362.
Nacelle + Air Induction	11297.
Tanks, TH Struct & Growth	24995.
Payload Supt & Separation	8300.
Total Structure	215243.
Engine, Thrust Rev + Exhaust	32160.
Starting + Control	632.
Fuel System	1823.
Rocket Propulsion	15663.
RCS Inerts	2103.
Total Propulsion	52380.
Flight Control	2666.
Auxiliary Power Plant	1626.
Instruments	1020.
Hydraulic, Pneumatic + Electric	10050.
Avionics	1998.
Furnishings + Equip	720.
Air Cond + Anti-Icing	1405.
Load + Handling	1520.
Total Fixed Equipment	21005.
Weight Empty	288629.
Crew	560.
Oil + Unusable Fuel	1208.
Non RCS WP & IFL	1951.
Residuals & Reserves @ LND	2367.
Non-Exp Useful Load	6086.
Operating Weight	294715.
Payload	577500.
Rocket Propellant-Ascent	299300.
Preignition Losses-Rocket	9955.
Fuel	128530.
GROSS WEIGHT	1310000.

Figure 3.39. CAV Weights

manufacturer's uninstalled performance data together with user supplied inlet, nozzle, and aftbody drag data.

Rocket engine performance is estimated using vacuum thrust and specific impulse corrected for ambient pressure effects.

3.6.5 Performance

The typical mission for the CAV consists of takeoff (with a ground roll of about 10,000 feet) and a climb to 30,000 ft and M = 0.862 using augmented, airbreathing engines.

After climbing to 30,000 ft and M = 0.862 under augmented airbreathing power, all rocket engines are ignited with the takeoff and climb taking 820.9 seconds. The vehicles proceed through a dual burn accelerated climb to the separation conditions under airbreathing and rocket thrust. During this initial boost the maximum dynamic pressure experienced is 1050 PSF and occurs at an altitude of 40,300 ft at M = 1.97. The vehicles separate at 117,500 ft, V = 3000 FPS where the dynamic pressure is 65 PSF. The orbiter proceeds to the required injection conditions for the particular mission with its propellant tanks full at separation. After separation the booster is lofted to 156,000 ft by its own momentum, descends, turns to the required heading, and returns to the launch site or an alternate base through powered and gliding flight. At no time does the booster fly faster than M = 2.95; experiences Mach numbers greater than 2.0 for only 212 seconds. This avoidance of a hostile flight environment enables the booster to be constructed of conventional materials without a thermal protection system. This is illustrated in Figure 3-40.

3.7 Hypersonic Interceptor - Model 1074-0006

3.7.1 Concept Description

The vehicle, illustrated in Figure 3.41, has an overall length of 169 ft 2 in and a wing span of 63 ft 4 in. The wing has a leading edge sweep of 72° on the inboard section and 50° on the outboard section, a reference area of 2,085 ft^2, an aspect ratio of 1.923, and a constant wing thickness ratio of 3.5%.

The airplane is designed for a one-man crew. Located forward, aft, and below the crew compartment are avionics/electronic equipment bays. Included in the 1,200 lbs of avionics equipment are target acquisition, communication, navigation and identification, information management, and defense functions. ECS equipment, oxygen, and electrical/hydraulic subsystem equipment are located in the

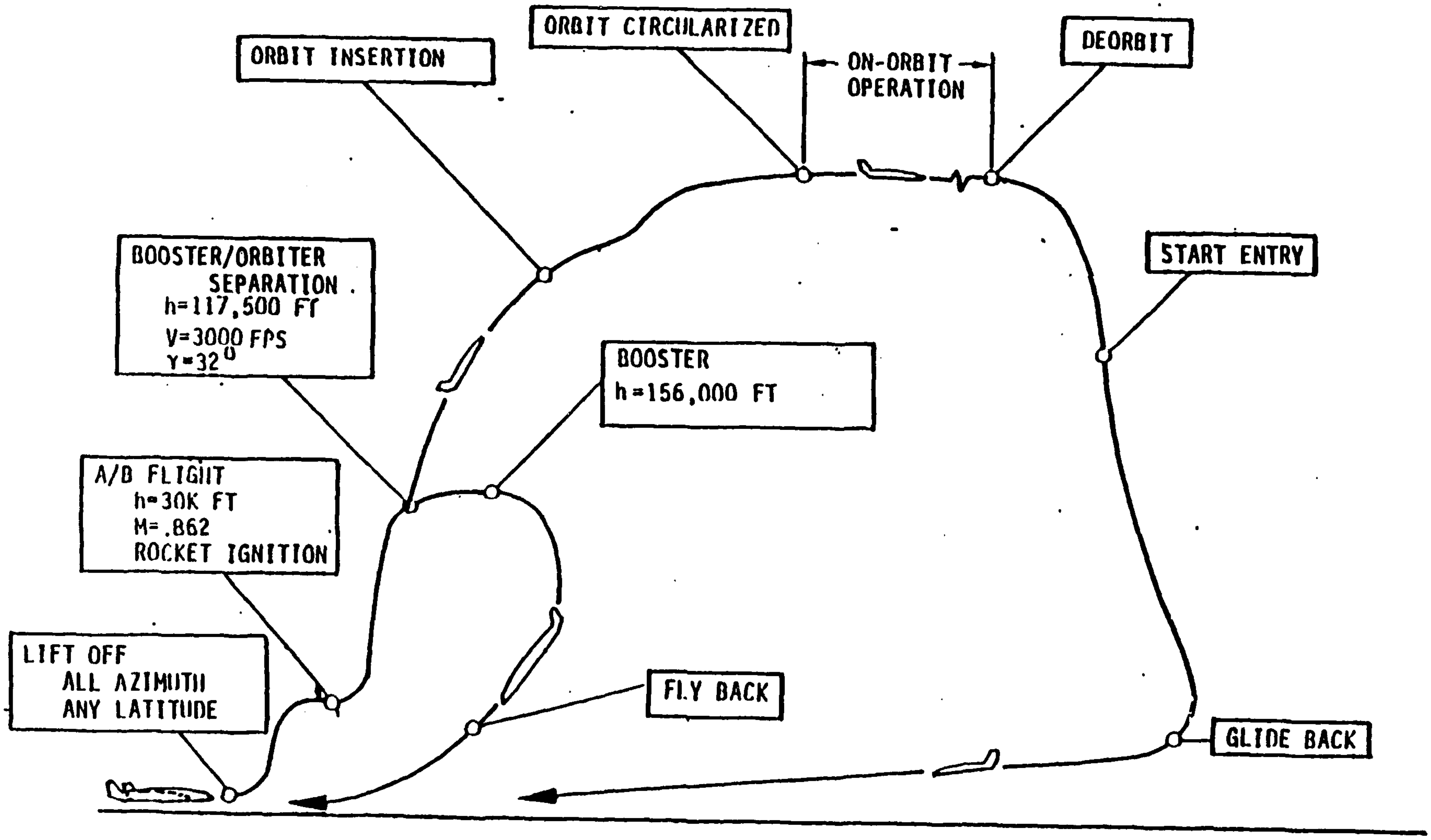


Figure 3-40. Boeing Modle 896-111 – Mission Profile

		WING	HORIZONTAL TAIL	VERTICAL TAIL
AREA	THEO SqFt	2085	—	—
	EXPOSED SqFt	855	289	240
SWEEP LE Deg		72	60	60
ASPECT RATIO		1.923	1.75	.98
TAPER RATIO	INBOARD	.201	.35	.39
	OUTBOARD	.565	—	—
SPAN Ft		63.33	22.5	15.33
THICKNESS t/c		.034	.05	.05
MAC	THEO In	550.75	—	—
	EXPOSED In	308.05	165.85	199.6
TAIL VOLUME		—	.1794	.1136

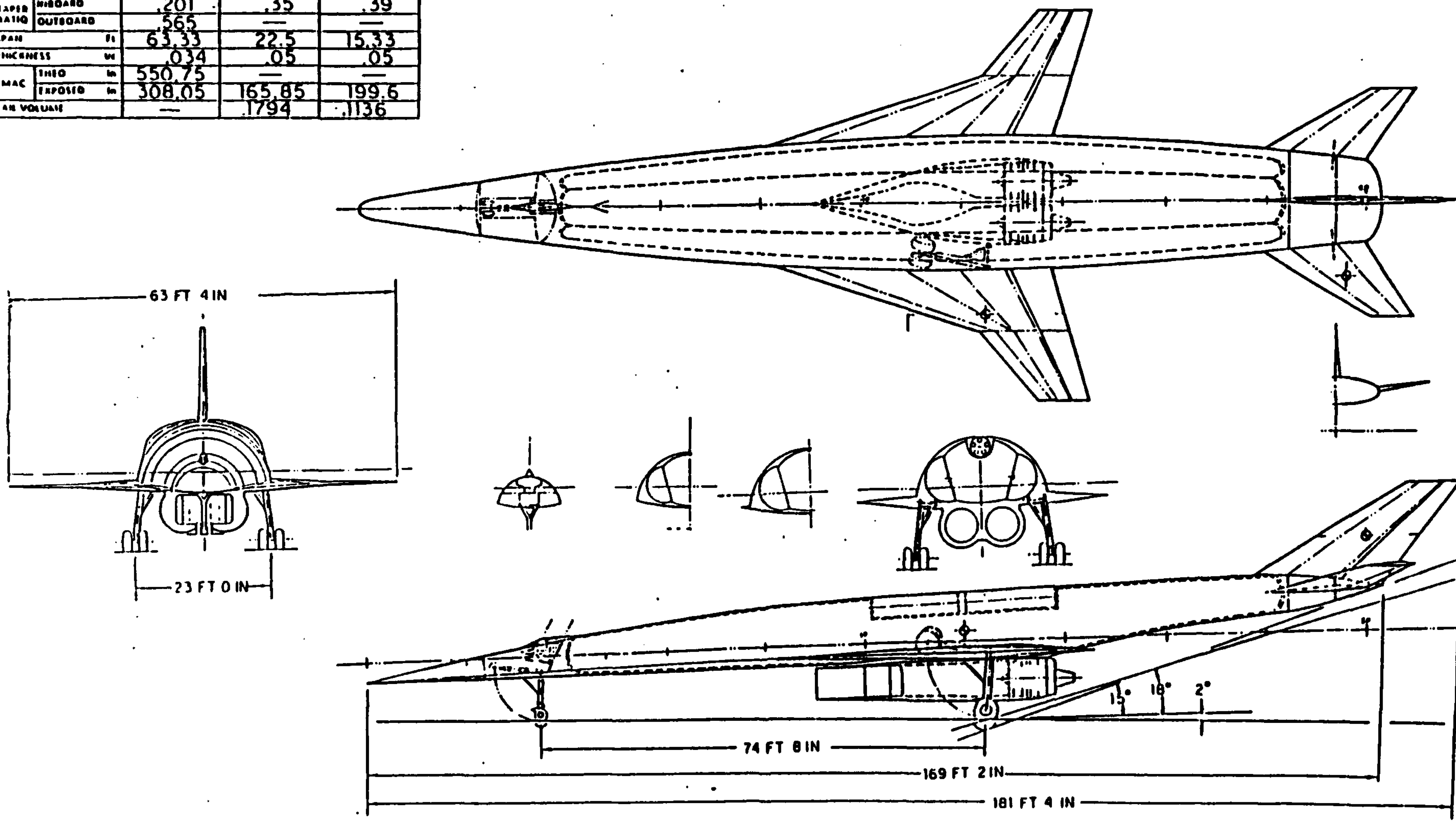


Figure 3-41. Hypersonic Interceptor Three-View Drawing

fuselage aft of the pilot. The body fuel is carried in integral insulated tanks with a capacity of 53,000 lbs of liquid hydrogen fuel.

The vertical fin has an area of 240 ft^2, a leading edge sweep of 60° and an aspect ratio of 0.98.

The horizontal tail has an area of 578 ft^2, leading edge sweep of 60° and an aspect ratio of 1.75.

3.7.2 Aerodynamic

Estimated aerodynamic characteristics of the Model 1074-0006 are presented in Figures 3-42 through 3-44.

Trimmed drag polars are shown for typical subsonic, supersonic and hypersonic flight conditions are shown in Figures 3-42 through 3-44.

In Figures 3-42 through 3-44, the drag generated by PWSIM is compared to LAAP (Large Airplane Analysis Program) and APAS (Aerodynamic Preliminary Analysis System) computer codes at subsonic and supersonic speeds and to APAS at hypersonic speeds.

3.7.3 Weights

The weight statement for the 1074-0006 is shown in Figure 3-45. Weight estimation ground rules and assumptions are listed below:

- The majority of aircraft structure is advanced hot structures, capable of enduring the high temperatures of sustained hypersonic flight
- Airframe Integrated Nozzle and Inlet
- Fly-by-wire surface controls
- Avionics equipment as described in Section 3.7.1
- Internal weapon carriage on two rotary launchers
- Final aircraft geometry is the result of aerodynamic and weight parametric trade studies and represents the best compromise for overall performance
- Judicious location of missiles and fuel such as to minimize CG travel as these items are expended
- Fuel pumping for trim control.

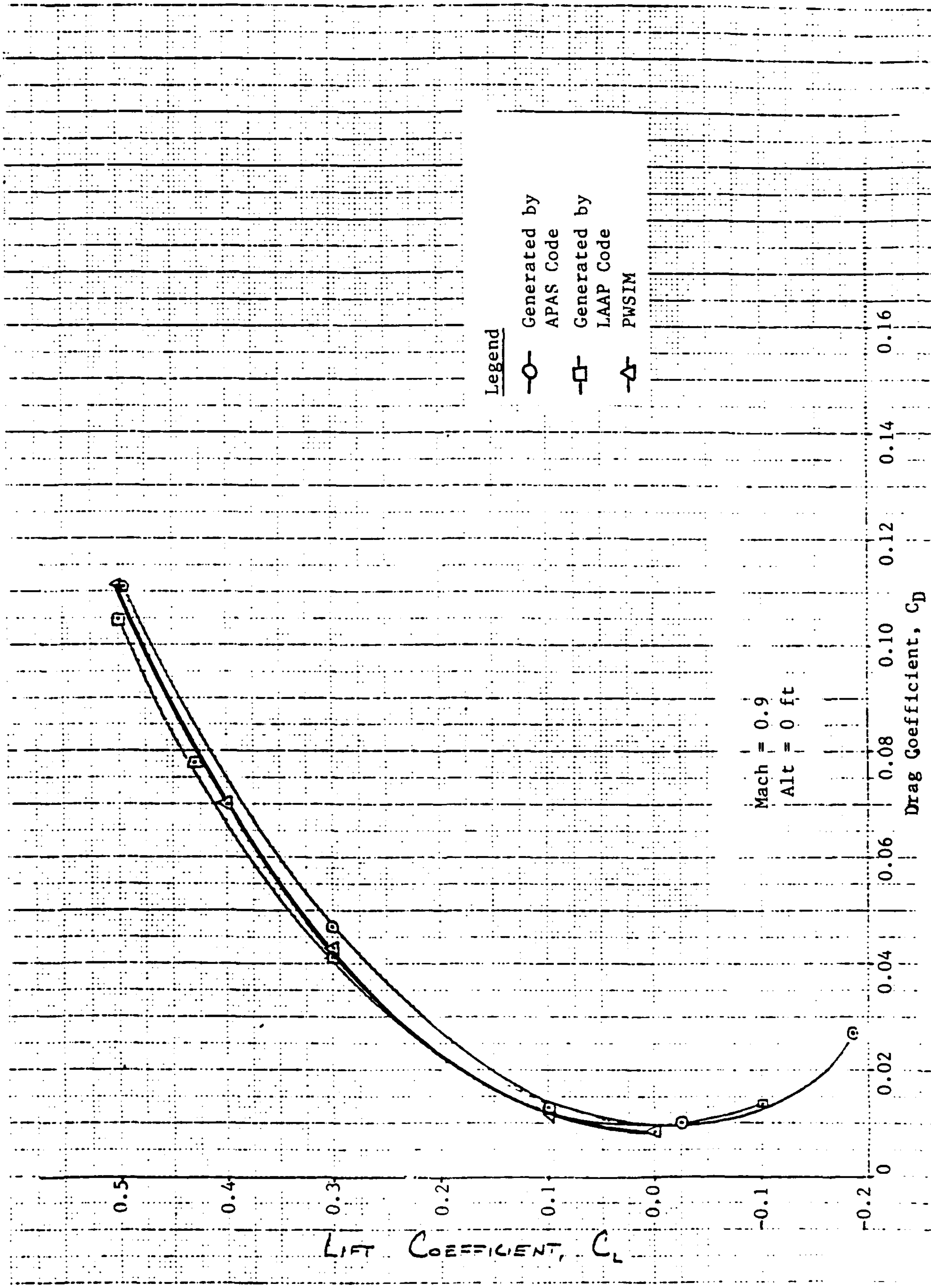


Figure 3-42. Hypersonic Interceptor Drag Polars (Subsonic)

CALC			REVISED	DATE
CHECK				
APR				
APR				

Model 1046-0006
Subsonic Drag Polars
(Computer Generated)

PAGE

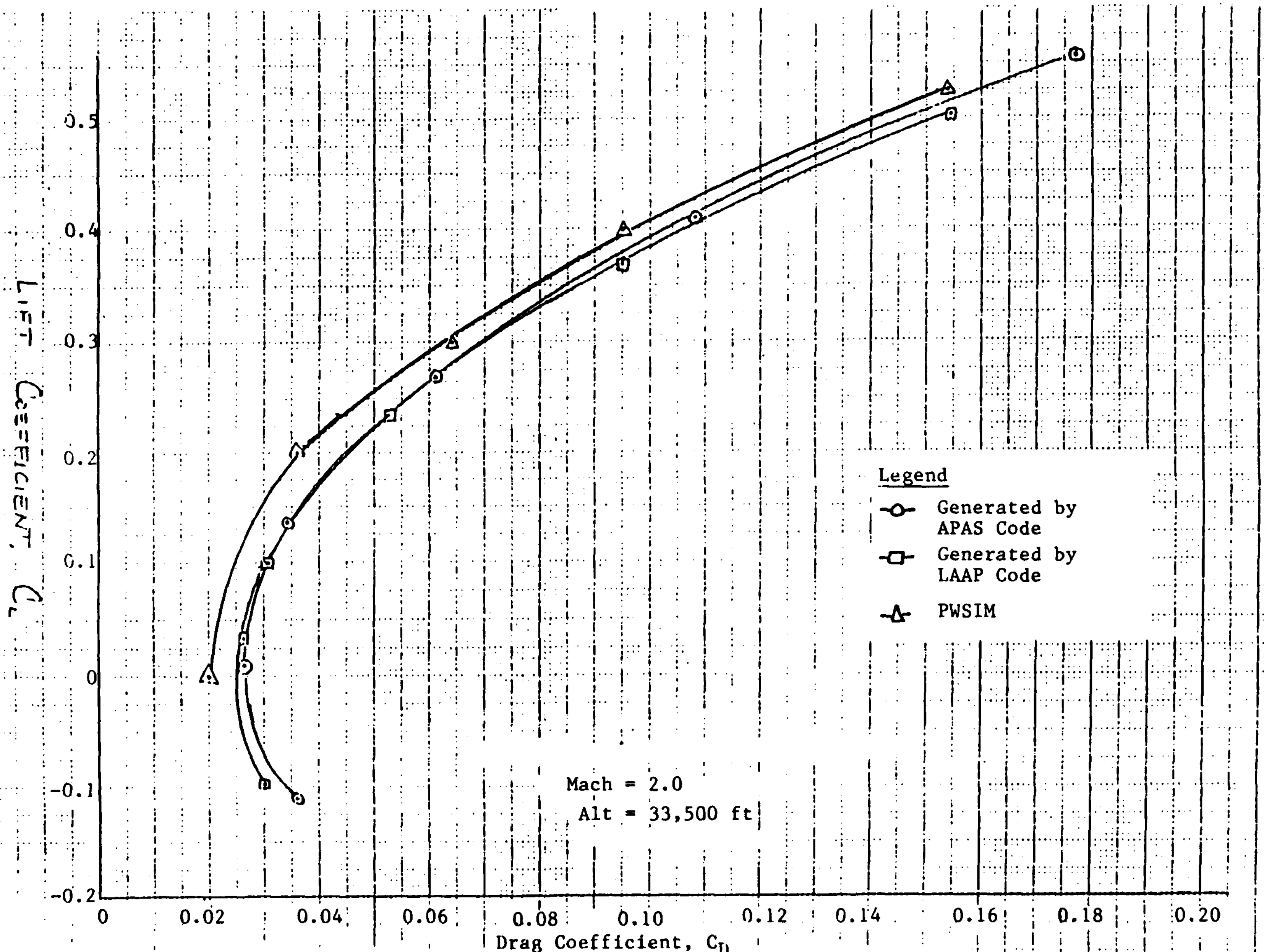


Figure 3-43. Hypersonic Interceptor Drag Polars (Supersonic)

MODEL 1046-0006
SUPERSONIC DRAG POLAR
(COMPUTER GENERATED)

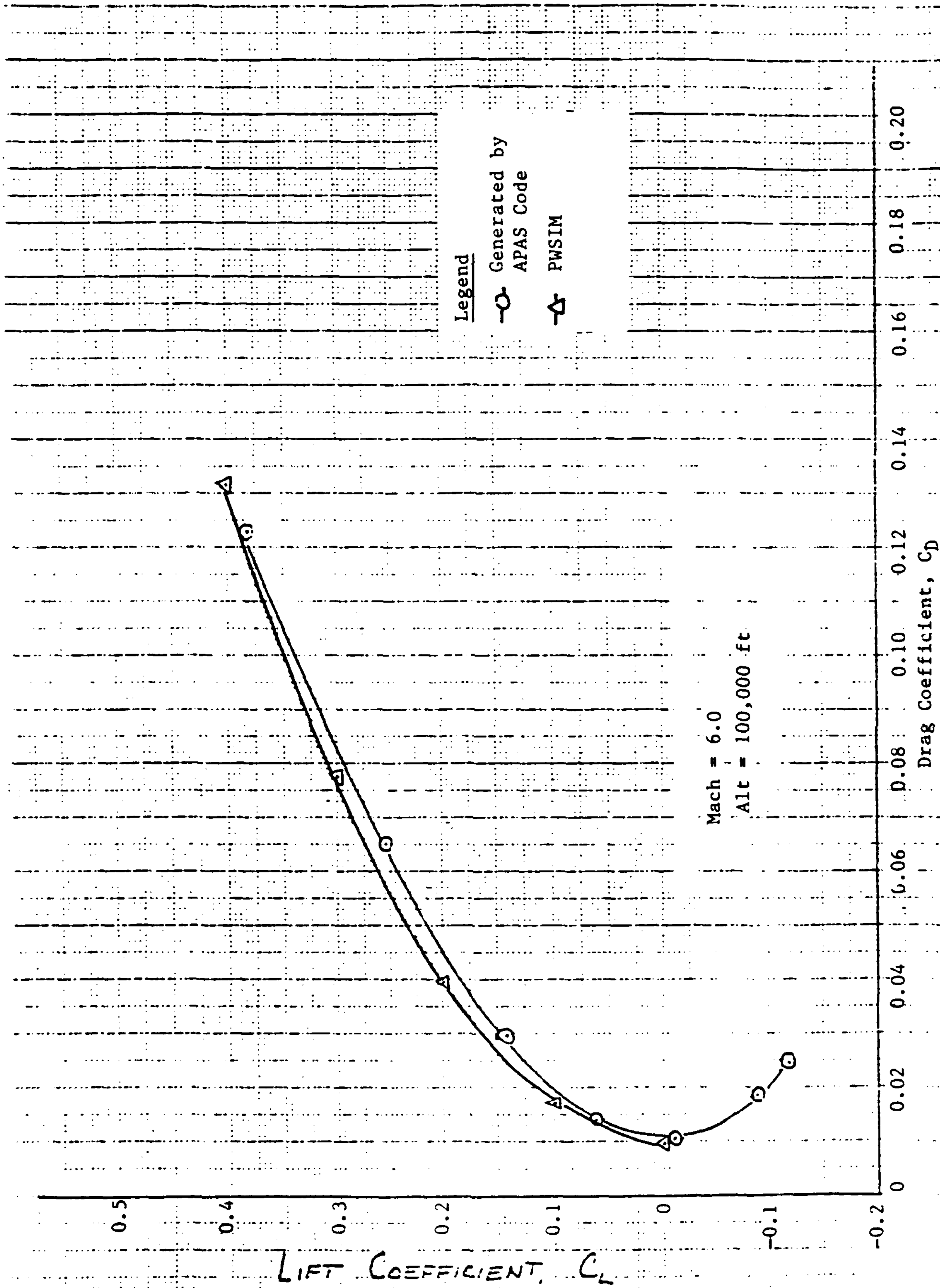


Figure 3-44. Hypersonic Interceptor Drag Polars (Hypersonic)

CALC			REVISED	DATE
CHECK				
APR				
APR				

MODEL 1046-0006
HYPERSONIC DRAG POLARS
(COMPUTER GENERATED)

PAGE

1074-0006 GROUP WEIGHT STATEMENT WTS 01-SEP-84 VERSION 87/02/17	WEIGHT-LBS
Wing	10128.
Tail	5267.
Body	23465.
Alighting Gear	4986.
Nacelle + Air Induction	5124.
Total Structure	48970.
Engine, Thrust Rev + Exhaust	7578.
Starting + Control	160.
Fuel System	2133.
Total Propulsion	9871.
Flight Control	1106.
Auxiliary Power Plant	500.
Instruments	220.
Hydraulic, Pneumatic + Electric	1846.
Avionics	1200.
Armament	340.
Furnishings + Equip	500.
Air Cond + Anti-Icing	855.
Load + Handling	20.
Total Fixed Equipment	6587.
Weight Empty	65427.
Crew + Equipment	280.
Oil + Unusable Fuel	1329.
Non-Exp Useful Load	1609.
Operating Weight	67037.
Payload	3000.
Fuel	52488.
GROSS WEIGHT	122525.

Figure 3-45. Weight Statement
Hypersonic Interceptor

3.7.4 Propulsion

Uninstalled engine performance was computed using the General Electric tandem turboramjet hyperjet, GE16/F40 study B1. The engine is a low bypass ratio, hydrogen fueled augmented turboramjet having a max augmented thrust of 57,718 lb sea level static. The engine cycle characteristics are bypass ratio (BPR) = 1.5, overall pressure ratio (OPR) = 25, turbine inlet temperature (T_{IT}) = STOICHIOMETRIC.

The inlet is located under the fuselage, centerline mounted. It is a two-dimensional, mixed compression inlet.

This inlet has a fixed first ramp, a flexible second ramp, and a movable third ramp. The boundary layer is controlled by means of porous bleed on the second and third ramp surfaces, sideplates, and a throat bleed slot located aft of the normal shock. The throat slot also acts as a bypass to remove excess inlet airflow for matching engine airflow demand with inlet supply and controls the position of the throat shock. The inlet capture area if 24.40 ft^2, sized for air requirements at Mach 5, 100,000 feet.

The aftbody of the interceptor serves as the expansion surface for the engine. Also, there is a turning vane which is used to maintain flow attachment of the exhaust plume on the aircraft aftbody throughout the aircraft flight regime.

3.7.5 Performance

The aircraft was configured to provide low drag at the design Mach number of 6.0. A design mission was specified (see Figure 3-46) that involved flight at altitudes greater than 100,000 feet, and sample results are shown in Figure 3-47.

4.0 Sample Results

This section contains an example PWSIM output (Figure 4-1). The output which is for the tactical fighter consists of:

- namelist inputs
- mission definitions
- airplane design (geometry) summary
- group weight statement
- weight design data and sensitivities

- detailed weights
- minimum profile drag
- wave drag
- drag due to lift
- zero lift drag versus Mach number
- mission results
- level flight performance
- engine data
- inlet tables
- aftbody drag tables
- installed engine performance
- airplane inlet maps
- airplane afterbody maps

1 TAKEOFF FUEL ALLOWANCE

- 2.5-MIN IDLE FUEL FLOW RATE
- 1/2-MIN MAX POWER FUEL FLOW RATE
- MAX POWER ACCELERATION TO CLIMB SPEED

2 MAX POWER CLIMB

3 HYPEROSNIC CRUISE TO INTERCEPT; OPTIMUM ALTITUDE

4 COMBAT

- (1) MAX POWER COMBAT TURN
- RELEASE PAYLOAD

5 HYPERSONIC CRUISE RETURN TO BASE

6 DECEL/DESCENT TO SEA LEVEL

7 RESERVES; 20-MIN SEA LEVEL LOITER, OPTIMUM MACH NUMBER

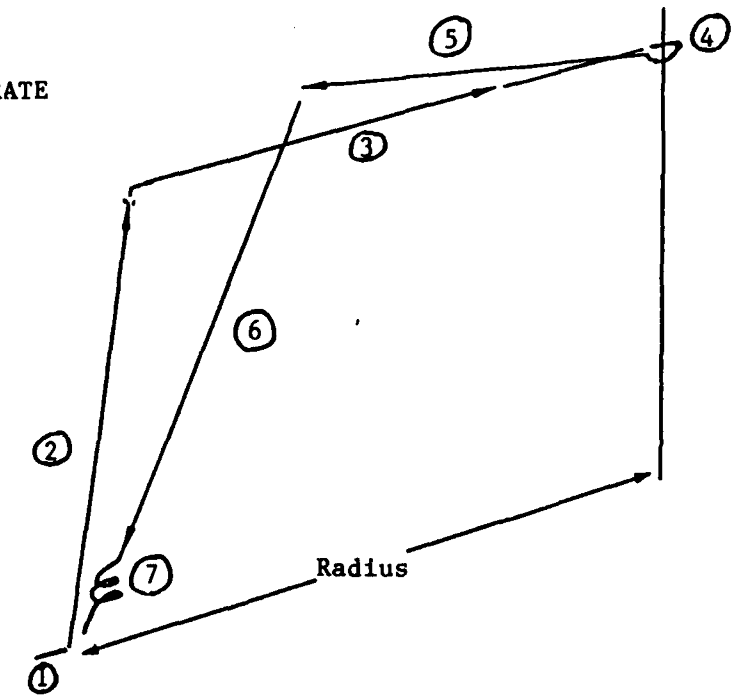


Figure 3-46. Design Mission Profile
Hypersonic Interceptor

MISSION: HYPMISS

NO	SEGMENT	PS	D	T	WTI	MI	ALTI	WTF	MF	ALTF	FUEL	WT
1	Taxi	.10	0	.042	122525	.001	0	122463	.001	0	61	122494
2	Taxi	2.00	0	.008	122463	.001	0	122021	.001	0	441	122242
3	Accel	2.00	3	.009	122021	.320	0	121489	.800	0	531	121755
4	Climb	2.00	884	.552	121489	.800	0	97267	6.019	104601	24222	109378
5	Cruise	1.49	2468	.697	97267	6.000	104687	86951	6.000	106858	10316	92109
6	Combat	1.39	0	.139	86951	6.000	85000	82242	6.000	85000	4708	84557
7	Drop	0	0	.000	82242	6.000	85000	82242	6.000	85000	0	
8	Cruise	1.49	3009	.844	82242	6.000	108120	71616	6.000	110639	10625	76929
9	Descent	.01	346	.295	71616	6.000	110650	70987	.800	0	628	71353
10	Loiter	.18	0	.333	70987	.352	0	70037	.349	0	950	70477

RADIUS = 3355 N.M.

Figure 3-47. Mission Summary Hypersonic Interceptor

```
$INPUTS
TITLE    = 'TEST CASE FOR BOEING MODEL 985-420, TACTICAL FIGHTER
OPTION   = 'DESIGN    ',
APTYPE   = 'TACFTR    ',
CKOUT    = 'LONG      ',
ENDJOB   = 'NO        ',
APNAME   = '985-420   ',  '          ',  '          ',
BRIEF    = 'YES       ',
FIXENG   = 'NO        ',
WNGFUEL  = 'YES       ',
TOGW     = .4E+05,
OEWA     = 0.0,
PAYLOAD  = .2E+04,
SCALE    = -.15E+01,
TOW      = .125E+01,
WOS      = .7E+02,
UVLF     = .12E+02,
ZMSUP    = .2E+01,
ZMSLM    = .12E+01,
CLMAX    = .15E+01,
CLMAXF   = .1E+01,
```

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

```
ALT      = .6E+05,
NPROP    = 77.
WPROP1   = .15E+01,
WDRIVE1  = .99E+05,
ZNE      = .2E+01,
FUELDEN  = .65E+01,
WEXTANK  = 0.0,
EXTFUEL  = 0.0,
BDOW     = .7387E+00,
BWHTF    = .7386E+00,
BWIDFC   = .1144E+01,
PF1      = .3094E+01,
PF2      = .4123E+01,
PF3      = .3356E+01,
XKAFT    = .5625E+00,
XKNOSE   = .4364E+00,
ZLDE     = .10177E+02,
RESFAC   = .1E+01,
AR       = .45E+01,
ETAIENG  = 0.0,
ETADENG  = 0.0,
TCR      = .5E-01,
TCS      = .5E-01,
TCT      = .35E-01,
TECHBOX  = .953E+00,
TR       = .25E+00,
XOIWING  = .5965E+00,
ZLAMLE   = .375E+02,
ARVT     = .13E+01,
TRVT     = .33E+00,
```

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

```
VBARV   = .7593E-01,
XOLVTL  = .9182E+00,
ZLAMVT  = .46E+02,
ARHT    = .2365E+01,
TRHT    = .2569E+00,
VBARH   = -.2634E+00,
XOLHTL  = .3713E+00,
ZLAMHT  = .45E+02,
DROTOR  = .1E+01,
LODINLT = .943E+01,
LODNOZ  = .162E+01,
RACAPT  = .527E+01,
RANOZZ  = .6E+01,
RDFF    = .2917E+01,
RDRF    = .3083E+01,
RENGWT  = .239E+04,
RLENG   = .9259E+01,
TSLS    = .25E+05,
SFCTKOF = .1E+01,
SFCTAXI = .1E+01,
SFCACCL = .1E+01,
SFCCLMB = .1E+01,
SFCCRUZ = .1E+01,
SFCCUMB = .1E+01,
SFCDESC = .1E+01,
SFCLOIT = .1E+01,
SFCREFU = .1E+01,
THRTKOF = .1E+01,
THRTAXI = .1E+01,
THRACCL = .1E+01,
```

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

```
THRCLMB = .1E+01,
THRCRUZ = .1E+01,
THRCOMB = .1E+01,
THRDESC = .1E+01,
THRLOIT = .1E+01,
THRREFU = .1E+01,
WOTHFE1 = .344E+03,
WOTHFE2 = 0.0,
WOTHPR1 = 0.0,
WOTHPR2 = 0.0,
WOTHST1 = 0.0,
WOTHST2 = 0.0,
WOTHUL1 = .39E+03,
WOTHUL2 = 0.0,
NOTHFE1 = 'RCS REDUCTION MAIL.         ',
NOTHFE2 = 'UNLABELED                   ',
NOTHPR1 = 'UNLABELED                   ',
NOTHPR2 = 'UNLABELED                   ',
NOTHST1 = 'UNLABELED                   ',
NOTHST2 = 'UNLABELED                   ',
NOTHUL1 = 'AMRAAM EJECTORS             '
NOTHUL2 = 'UNLABELED                   ',
DFSUB   = 0.0,
FACDRG  = .1E+01,
E       = .8E+00,
NFACW   = 4,
XMCDW   = .1E+01, .12E+01, .16E+01, .2E+01, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
CDWFAC1 = .917E+00, .849E+00, .89E+00, .1105E+01, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
NSTRMCH = 9,
STORMCH = 0.0, .85E+00, .9E+00, .1E+01, .11E+01, .125E+01, .15E+01, .2E+01,     .26E+01, 0.0,
```

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

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STORDF  = .3E+00, .3E+00, .34E+00, .5E+00, .47E+00, .42E+00, .4E+00, .38E+00,   .38E+00, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
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          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
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XMTAB   = 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
NXHTAB  = 0,
XHTAB   = 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
CDOTAB  = 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
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          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
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NCLTAB  = 0,
CLTAB   = 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
CDITAB  = 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0, 0.0,
          0.0,
MISSION = 'SUBSUB  ', '          ', '          ', '          ', '          ', '          ', '          ', '          ', '
GUES    = .1E+04, -.1E+01, -.1E+01, -.1E+01, -.1E+01, -.1E+01, -.1E+01,        -.1E+01, -.1E+01, -.1E+01,
$END
       MISSIONS          GUES
       **********     **********

NAME = SUBSUB     AND GUESS =    1000.00
```

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

START OF PROCESSING FOR MISSION SUBSUB

SEGMENT	PS	EXTENT	MACH I	ALT I	MACH F	ALT F
TAXI	2.	.00833	0.01	0.	0.25	0.
TAXI	0.01	.08333	0.01	0		
ACCEL	2.		0.25	0	0.85	0.
CLIMB	1.		0.85	0.	0.85	NEXT
CRUISE	1.		0.9	OPT		
RADIUS						
LOITER	1.	1.	0.69	36295.	0.69	OPT
COMBAT	2.	1.	0.8	10000.	0.8	10000.
DROP	2000.					
CLIMB	1.		0.85	10000.	0.85	NEXT
CRUISE	1.		0.9	OPT		
LOITER	1.	0.33333	0.27	0.		

START OF PROCESSING FOR MISSION SUBSUB2

SEGMENT	PS	EXTENT	MACH I	ALT I	MACH F	ALT F
DROP	-2000.	1.				
TAXI	2.	.00833	0.01	0	0.25	0.
ACCEL	2.		0.25	0	NEXT	0.
CLIMB	1.		0.85	0	NEXT	NEXT
CRUISE	1.		OPT	OPT		
RADIUS						
LOITER	1.	1.	OPT	OPT		
COMBAT	2.	1.	0.8	10000.	0.8	10000.
DROP	2000.	0.				
CLIMB	1.		0.85	10000.	NEXT	NEXT
CRUISE	1.		OPT	OPT		
LOITER	1.	0.33333	OPT	0.		0.

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

AIRPLANE DESIGN SUMMARY

DATE	: 85/12/12.
TIME	: 23.18.06.
AIRCRAFT TYPE	: TACFIR
DERIVED FROM BASELINE	: 985-420

TAKEOFF GROSS WEIGHT	:	40000.	POUNDS
WING LOADING	:	70.00	LB/SQ.FT
THRUST / WEIGHT RATIO	:	1.2500	

ENGINE IDENTIFICATION :
:::::::: TI-FI JI69621 CONFIG.3:SUBSONIC/SUPERSONIC DASH CASE1 CARD2E

ENGINE SCALE	:	1.0000
LENGTH (FT)	:	9.26
DIAMETER (FT)	:	3.08
CAPTURE AREA (SQ.FT.)	:	5.67
NOZZLE AREA (SQ.FT.)	:	6.00

		WING	HORIZ. TAIL	VERT. TAIL
AREA (REFERENCE)	SQ.FT	571.43	136.94	55.52
AREA (EXPOSED)	SQ.FT	447.76	79.29	55.52
AREA (WETTED)	SQ.FT.	895.52	158.58	222.07
SWEEP (L.E.)	DEGREES	37.50	45.00	46.00
SWEEP (C/4)	DEGREES	32.37	36.87	40.09
SWEEP (C/2)	DEGREES	26.60	26.57	32.94
ASPECT RATIO		4.500	2.365	1.300
TAPER RATIO		.250	.257	.330
THICKNESS/CHORD RATIO		.0470	.0300	.0300
MEAN AERO CHORD	FEET	12.62	6.27	7.09
SPAN	FEET	50.71	18.00	8.50
TAIL VOLUME COEFFICIENT			-.2634	.0759

		BODY	NACELLE	GEAR POD
LENGTH	FEET	61.60	.00	.00
WIDTH	FEET	7.25	.00	
DEPTH	FEET	5.35		
FINENESS RATIO (L/D)		10.177	.000	.000
WETTED AREA (AERO)	SQ.FT.	1012.10	.00	.00
WETTED AREA (STRUC)	SQ.FT.	1030.24		
FRONTAL AREA	SQ.FT	28.77	.00	.00
VOLUME	CU.FT.	1045.10		
TOTAL WETTED AREA	SQ.FT.	2288.27		

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

985-420 GROUP WEIGHT STATEMENT WTS 01-SEP-84 VERSION 85/12/12.	WEIGHT-LBS
WING	3802.
TAIL	1228.
BODY	5092.
ALIGHTING GEAR	1711.
NACELLE + AIR INDUCTION	1148.
TOTAL STRUCTURE	12980.
ENGINE, THRUST REV + EXHAUST	4961.
STARTING + CONTROL	160.
FUEL SYSTEM	704.
TOTAL PROPULSION	5825.
FLIGHT CONTROL	945.
AUXILIARY POWER PLANT	300.
INSTRUMENTS	160.
HYDRAULIC, PNEUMATIC + ELECTRIC	953.
AVIONICS	1859.
ARMAMENT	340.
FURNISHINGS + EQUIP	225.
AIR COND + ANTI ICING	797.
LOAD + HANDLING	10.
RCS REDUCTION MATL.	344.
TOTAL FIXED EQUIPMENT	5933.
WEIGHT EMPTY	24739.
CREW + EQUIPMENT	280.
OIL + UNUSABLE FUEL	306.
PAYLOAD INST + WEAPONS	685.
AMRAAM EJECTORS	390.
NON-EXP USEFUL LOAD	1661.
OPERATING WEIGHT	26400.
PAYLOAD	2000.
FUEL	11600.
GROSS WEIGHT	40000.

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

```
                    985-420
                    DESIGN DATA AND SENSITIVITIES
                    85/12/12.

GROSS WT, LB               40000.     V STALL, KNOTS             107.
FLIGHT DESIGN WT, LB       37680.     CLMAX-LDG                    1.50
ULT VERT LOAD FACTOR          12.00   STRUCTURE/GW                  .325
LANDING WT, LB             33040.     PROPULSION/GW                 .146
OPERATING WT, LB           26400.     FIXED EQUIP/GW                .148
WEIGHT EMPTY, LB           24739.     NON-EXP USEFUL LOAD           .042
AIRFRAME UNIT WT, LB       16637.     OW/GW                         .660
ALTITUDE, FT               60000.     PAYLOAD/GW                    .050
MACH MAX                       2.00   EXP USEFUL LOAD/GW            .000
MACH SL                        1.20   FUEL/GW                       .290
Q MAX, PSF                  2133.
    WING-TRAP
AREA GROSS, SQ FT            571.4    SWEEP LE, DEG               37.5
AREA EXPOSED, SQ FT          447.8    SWEEP EA, DEG               26.6
SPAN, FT                      50.7    MAC, FT                     12.6
ASPECT RATIO                   4.22   UNIT WT SG, PSF              6.65
TAPER RATIO                     .25   UNIT WT SE, PSF              8.49
T/C ROOT                        .050  WING LOAD GW, PSF           70.
T/C SOB                         .050  WING LOAD UDW, PSF         791.
T/C TIP                         .035
    H-TAIL TRAP
AREA GROSS, SQ FT            136.9    SWEEP C/2, DEG              26.6
AREA EXPOSED, SQ FT           79.3    TAIL ARM, FT               -13.9
SPAN, FT                      18.0    VOLUME COFF                  -.26
ASPECT RATIO                   2.37   PERCENT ELEVATOR              0
TAPER RATIO                     .26   PITCH ACC, RAD/SEC           6.0
T/C ROOT                        .030  UNIT WT SG, PSF              4.06
T/C SOB                         .030  UNIT WT SE, PSF              7.01
T/C TIP                         .030  TAIL LOAD, PSF            1087.
SWEEP LE, DEG                 45.0
    V-TAIL(2 )
AREA, SQ FT                   55.5    SWEEP C/2, DEG              32.9
SPAN, FT                       8.5    TAIL ARM, FT                19.8
ASPECT RATIO                   1.30   VOLUME COFF                   .076
TAPER RATIO                     .33   PERCENT RUDDER              30.0
T/C ROOT                        .030  UNIT WT, PSF                 6.05
T/C TIP                         .030  TAIL LOAD, PSF             332.
SWEEP LE, DEG                 46.0
    BODY
WETTED AREA, SQ FT          1030.2    LENGTH/DEPTH                11.5
LENGTH, FT                    61.6    DELTA P, PSI                  .0
MAX WIDTH, FT                  7.25   UNIT WT, PSF                 4.94
MAX DEPTH, FT                  5.35
    LANDING GEAR
LG WT/LANDING WT                .052  LANDING KE, K FT-LB     15892.
    PROPULSION
SLST PER ENG, LB(2 )       25000.     SLST/GW                      1.25
WING FUEL, GAL               406.6    SLST/ENG WT                 10.46
BODY FUEL, GAL              1378.0
    SYSTEMS
VOLUME PRES, CU FT            70.     CREW                         1.
```

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

```
DETAIL WEIGHTS AND PARAMETERS                    PAGE 1 OF 2
    NOTE:  THESE DATA ARE BACKUP; USE WITH CAUTION.

--STRUCTURE------------------------------------------------------
WWING  =   3801.534                   WBODY   =   5092.278
  WBOX    =   2767.258                  WBDSIRC=   4012.549
    WBASBOX=   2373.072                   WBASBOD=    4044.406
    KTOCW  =      1.747                   KBPRES  =      1.000
    KARW   =       .619                   KBGEAR  =      1.100
    KSWPWEA=      1.042                   KWEP    =      1.030
    KTEMPBX=       .751                   KBENG   =      1.060
    KWGEAR =      1.000                   KLFNDP  =      1.038
    KDW    =       .998                   KFLOOR  =      1.000
    KNWS   =      1.449                   KTEMPB  =       .796
    TECHBOX=       .953                 WRAMP   =       .000
  WCONSUR=    619.625                   WDOOR   =       .000
    KTEMPCS=       .744                 WMIDWNG=    969.728
    TECHCS =      1.000                   KMIDWNG=       .350
  WMISCW =    414.652                   WGRPOD  =       .000
    KTEMPM =       .760                 TECHBOD=      1.000
    TECHMIS=      1.000                 KBODY   =      1.000
  KWING  =      1.000                   WALGR   =   1711.406
WHTAIL =    555.538                     KCBR    =      1.104
  WHTSIRC=    523.266                   TECHAG  =       .913
    WHBAS  =    226.702
    KLOADHT=      2.452                 KALGR   =      1.000
    KSTAB  =      1.250
    KTEMPHT=       .753               WNAC   =    236.009
  WELEVIR=       .000                   WENGMIS=     75.000
  WHTAITH=     32.272                   WFIREWL=     29.893
    KITE   =      1.000                 NENG    =      2.000
  TECHHT =      1.000                   WCOWL   =       .000
                                          WNCOWL  =       .000
  KHTAIL =      1.000                     WFCOWL  =       .000
                                          WSCOWL  =       .000
WVTAIL =    672.053                     WSTRUT  =       .000
  WVTSIRC=    261.579                   NSTRUT  =       .000
    WVTBAS =    275.227                 KTEMPN  =      1.125
    KLOADVT=      1.267                 TECHNAC=      1.000
    KVTDES =      1.000
    KTEMPVT=       .750                 KNAC    =      1.000
  WRUDDER=     74.448
  TECHVT =      1.000                 WAIRIN =    911.568
  NVTAIL =      2.000                   WINLET  =    368.079
                                        WDUCT   =    206.680
  KVTAIL =      1.000                   KTEMPIN=       .793
                                        TECHNAC=      1.000
                                        KAIRIN  =      1.000
                                        NENG    =      2.000
                                      WDIHST1=       .000
                                      WDIHST2=       .000
```

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

```
DETAIL WEIGHTS AND PARAMETERS                      PAGE 2 OF 2

--PROPULSION----------------------------------------------------
WENG    =  4961.000                     WFUSYS =    704.155
  WENG1   =  2390.000                     WFSBAS =    714.371
  NENG    =     2.000
WTREV   =      .000                       TECHFUL=       .980
  WTREV1  =      .000                     KFUSYS =       .841
WEXH    =      .000                     WDRIVE =       .000
  WEXHI   =      .000                     WDRIVE1= 99000.000
WESTART=      .000                        NPROP  =       .000
WENGCON=   160.000                      WPROPLL=       .000
  WTHROTL=    30.000                      WPROP1 =      1.500
  WTRCON =      .000                    WOTHPR1=       .000
                                        WOTHPR2=       .000

--FIXED EQUIPMENT-----------------------------------------------
WFLTCON=   945.361                      WFURN  =    225.000
  WCOCKP =    33.000                      WPERSNL=    170.000
  WAUTOP =   141.667                      WMISCF =     30.000
  WROLCON=   287.615                      WFUR   =     15.000
  WPTICON=   338.470                      WEMERG =     10.000
  WRUDCON=    84.112                      TECHFUR=      1.000
  WHLCON =   292.878
  WBRKCON=    50.000                      KFURN  =      1.000
  TECHFLT=      .770
                                        WAIRCON=    729.923
  KFLTCON=     1.000                      WAC    =    768.340
                                          TECHAC =       .950
WAPU    =   300.481
WINSTR  =   160.000                       KAIRCON=      1.000
  WFLGTIN=    50.000
  WENGIN =    60.000                    WANTICE=     67.067
  WMISIN =    50.000                      WWSICE =     20.000
                                          WAIRICE=     50.596
  KINSTR =     1.000                      WPRPICE=       .000
                                          WWNGICE=       .000
WHYD    =   419.885                       TECHAC =       .950
  KMACH  =     1.516
  TECHHYD=      .940                      KANTICE=      1.000
  KHYD   =     1.000                    WLDHDLG=     10.000
                                          WHDLG  =     10.000
WELEC   =   532.705                       WLOAD  =       .000
  TECHEL =      .800                      TECHLDH=      1.000
  KELEC  =     1.000                      KLDHDLG=      1.000
WAVION  =  1859.000                     WOTHFE1=    344.000
WARM    =   340.000                     WOTHFE2=       .000

--NON-EXP USEFUL LOAD-------------------------------------------
WCREW   =   230.000                     WPLPROV=       .000
WPROV   =    50.000                     WGUN   =    685.000
WOIL    =   190.000                     WOTHUL1=    390.000
WUNAFU  =   116.022                     WOTHUL2=       .000
```

Figure 4-1. TAPE6 — General Aircraft Output Data (Continued)

```
MINIMUM PROFILE DRAG

          MACH          =2.000
          ALT           =45000.
          RE/FT'10-6  =   3.01

          S REF         =  571.
          AWET          = 2288.

                                       FLAT PLATE        BASIC
               GEOMETRY                SKIN FRICTION     PROFILE DRAG

                                 T/C OR   RE'
COMPONENT     A WET  LENGTH  L/D      10-6     CF       CDF      K        CDBP

 WING          896.   12.6  .0425    38.0   .00177   .00277  1.047   .00291
 BODY         1012.   61.6  10.18   185.6   .00140   .00248  1.053   .00261
 H TAIL        159.    6.3  .0300    18.9   .00198   .00055  1.065   .00058
 V TAIL        222.    7.1  .0300    21.4   .00194   .00075  1.030   .00078

 TOTAL AWET   2288.                TOTAL CD BASIC PROFILE =   .00688

                          PRESSURE DRAG

 BODY                                                 CD =   .00020
 NACELLE                                              CD =   .00000

                                           TOTAL CD PRESS=   .00020

                          INTERFERENCE DRAG

 WING/BODY                                       CD INTF =   .00000
 V-TAIL/BODY                                     CD INTF =   .00000
 H-TAIL/BODY( OR V TAIL)                         CD INTF =   .00000
 WING/STRUT/NACELLE                              CD INTF =   .00000

                                            TOTAL CDINTF =   .00000

                                     CD PRESS + CD INTF =   .00020

                          CD BPRO + CD PRESS + CD INTF =   .00708

   K EXCR =    .0829                             CD EXCR =   .00094

                              SUB TOTAL CD BPRO + CD EXCR =   .00781

      KO= ( CDPRESS + CD INTF)/( CD BPRO + CD EXCR ) =   .02560
         ***   TOTAL   CDP MIN   ****                     =   .00801
```

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

CDPMIN TABLE VS. MACH NO. AND ALTITUDE

ALT-FT	0.	15000.	30000.	45000.	60000.	75000.	90000.
M= .1000	.013052	.013970	.015104	.016830	.019127	.021966	.025394
M= .3000	.010984	.011703	.012588	.013921	.015674	.017812	.020355
M= .5000	.010064	.010704	.011489	.012668	.014211	.016081	.018292
M= .7000	.009402	.009990	.010711	.011789	.013196	.014897	.016900
M= .9000	.008940	.009494	.010172	.011184	.012501	.014090	.015957
M=1.1000	.008922	.009471	.010142	.011142	.012441	.014005	.015842
M=1.2000	.008572	.009098	.009741	.010697	.011939	.013434	.015188
M=1.3000	.008272	.008779	.009398	.010319	.011514	.012951	.014636
M=1.4000	.007981	.008469	.009066	.009953	.011103	.012485	.014106
M=1.5000	.007698	.008169	.008744	.009598	.010705	.012035	.013594
M=1.6000	.007424	.007878	.008432	.009255	.010321	.011601	.013102
M=1.7000	.007163	.007601	.008136	.008929	.009956	.011190	.012635
M=1.8000	.006910	.007332	.007849	.008614	.009604	.010792	.012185
M=1.9000	.006665	.007072	.007571	.008309	.009263	.010408	.011751
M=2.0000	.006427	.006821	.007302	.008014	.008933	.010038	.011331

****** COMPONENT WAVE DRAG COEFFICIENTS ******

MACH	BODY	WINGS	TAILS	NACELLE	TOTAL
.100	.00000	.00000	.00000	.00000	.00000
.300	.00000	.00000	.00000	.00000	.00000
.500	.00000	.00000	.00000	.00000	.00000
.700	.00000	.00000	.00000	.00000	.00000
.900	.00000	.00000	.00000	.00000	.00000
1.100	.00458	.01352	.00106	.00000	.01916
1.200	.00454	.01516	.00127	.00000	.02098
1.300	.00459	.01566	.00148	.00000	.02173
1.400	.00465	.01467	.00150	.00000	.02082
1.500	.00470	.01365	.00152	.00000	.01987
1.600	.00476	.01260	.00153	.00000	.01889
1.700	.00505	.01208	.00163	.00000	.01876
1.800	.00533	.01142	.00172	.00000	.01847
1.900	.00562	.01061	.00181	.00000	.01804
2.000	.00591	.00965	.00190	.00000	.01747

DRAG-DUE-TO-LIFT FACTORS (CD(LIFT) = K1(CL**2) + K2(CL**4))

MACH NO	XKL1	CDLFACT	K1	XKL2	CDLFAC2	K2
.10000	.06349	1.00000	.06349	.05038	1.00000	.05038
.30000	.06550	1.00000	.06550	.04711	1.00000	.04711
.50000	.06654	1.00000	.06654	.04346	1.00000	.04346
.70000	.06841	1.00000	.06841	.04114	1.00000	.04114
.90000	.08188	1.00000	.08188	.04402	1.00000	.04402
1.10000	.11017	1.00000	.11017	.19570	1.00000	.19570
1.20000	.15494	1.00000	.15494	.21826	1.00000	.21826
1.30000	.19459	1.00000	.19459	.18184	1.00000	.18184
1.40000	.23215	1.00000	.23215	.11759	1.00000	.11759
1.50000	.27230	1.00000	.27230	.07582	1.00000	.07582
1.60000	.30664	1.00000	.30664	.06698	1.00000	.06698
1.70000	.34010	1.00000	.34010	.05499	1.00000	.05499
1.80000	.37298	1.00000	.37298	.04010	1.00000	.04010
1.90000	.40534	1.00000	.40534	.03663	1.00000	.03663
2.00000	.43739	1.00000	.43739	.03928	1.00000	.03928

Figure 4-1. TAPE6 — General Aircraft Output Data (Continued)

CDPMIN TABLE VS. MACH NO. AND ALTITUDE

ALT-FT	0.	15000.	30000.	45000.	60000.	75000.	90000.
M= .1000	.013052	.013970	.015104	.016830	.019127	.021966	.025394
M= .3000	.010984	.011703	.012588	.013921	.015674	.017812	.020355
M= .5000	.010064	.010704	.011489	.012668	.014211	.016081	.018292
M= .7000	.009402	.009990	.010711	.011789	.013196	.014897	.016900
M= .9000	.008940	.009494	.010172	.011184	.012501	.014090	.015957
M=1.1000	.008922	.009471	.010142	.011142	.012441	.014005	.015842
M=1.2000	.008572	.009098	.009741	.010697	.011939	.013434	.015188
M=1.3000	.008272	.008779	.009398	.010319	.011514	.012951	.014636
M=1.4000	.007981	.008469	.009066	.009953	.011103	.012485	.014106
M=1.5000	.007698	.008169	.008744	.009598	.010705	.012035	.013594
M=1.6000	.007424	.007878	.008432	.009255	.010321	.011601	.013102
M=1.7000	.007163	.007601	.008136	.008929	.009956	.011190	.012635
M=1.8000	.006910	.007332	.007849	.008614	.009604	.010792	.012185
M=1.9000	.006665	.007072	.007571	.008309	.009263	.010408	.011751
M=2.0000	.006427	.006821	.007302	.008014	.008933	.010038	.011331

****** COMPONENT WAVE DRAG COEFFICIENTS ******

MACH	BODY	WINGS	TAILS	NACELLE	TOTAL
.100	.00000	.00000	.00000	.00000	.00000
.300	.00000	.00000	.00000	.00000	.00000
.500	.00000	.00000	.00000	.00000	.00000
.700	.00000	.00000	.00000	.00000	.00000
.900	.00000	.00000	.00000	.00000	.00000
1.100	.00458	.01352	.00106	.00000	.01916
1.200	.00454	.01516	.00127	.00000	.02098
1.300	.00459	.01566	.00148	.00000	.02173
1.400	.00465	.01467	.00150	.00000	.02082
1.500	.00470	.01365	.00152	.00000	.01987
1.600	.00476	.01260	.00153	.00000	.01889
1.700	.00505	.01208	.00163	.00000	.01876
1.800	.00533	.01142	.00172	.00000	.01847
1.900	.00562	.01061	.00181	.00000	.01804
2.000	.00591	.00965	.00190	.00000	.01747

DRAG-DUE-TO-LIFT FACTORS (CD(LIFT) = K1(CL**2) + K2(CL**4))

MACH NO	XKL1	CDLFAC1	K1	XKL2	CDLFAC2	K2
.10000	.06349	1.00000	.06349	.05038	1.00000	.05038
.30000	.06550	1.00000	.06550	.04711	1.00000	.04711
.50000	.06654	1.00000	.06654	.04346	1.00000	.04346
.70000	.06841	1.00000	.06841	.04114	1.00000	.04114
.90000	.08188	1.00000	.08188	.04402	1.00000	.04402
1.10000	.11017	1.00000	.11017	.19570	1.00000	.19570
1.20000	.15494	1.00000	.15494	.21826	1.00000	.21826
1.30000	.19459	1.00000	.19459	.18184	1.00000	.18184
1.40000	.23215	1.00000	.23215	.11759	1.00000	.11759
1.50000	.27230	1.00000	.27230	.07582	1.00000	.07582
1.60000	.30664	1.00000	.30664	.06698	1.00000	.06698
1.70000	.34010	1.00000	.34010	.05499	1.00000	.05499
1.80000	.37298	1.00000	.37298	.04010	1.00000	.04010
1.90000	.40534	1.00000	.40534	.03663	1.00000	.03663
2.00000	.43739	1.00000	.43739	.03928	1.00000	.03928

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

OTHER DRAG CONTRIBUTIONS :

INLET(REF)		NOZZLE(REF)
MACH	CD(INLET)	CD(NOZZLE)
.000	.00000	-.00062
.300	.00000	-.00068
.600	.00002	-.00077
.601	.00001	-.00094
.800	.00027	-.00096
.900	.00043	-.00133
1.000	.00057	-.00658
1.200	.00071	-.00658
1.400	.00109	-.00725
1.600	.00165	-.00795
1.800	.00161	-.00810
2.000	.00117	-.00813

DIVERTER	
MACH	CD(DIV)
.700	.00000
.800	.00003
.900	.00008
1.000	.00015
1.100	.00017
1.200	.00018
1.300	.00018
1.400	.00018
1.500	.00019
1.600	.00019
1.700	.00019
1.800	.00019
1.900	.00020
2.000	.00020
2.100	.00020
2.200	.00020

TOTAL CD = CD(PROFILE) + CD(WAVE) + CD(INLET) + CD(NOZZLE) + CD(DIV) + CD(LIFT) + CD(STORE)

NOTE : EXTERNAL STORE DRAG IS APPLIED IN SUBROUTINE DRAG AS REQUIRED DURING MISSION SEGMENT CALCULATIONS

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

COEFFICIENTS OF DRAG AT ZERO LIFT
(NO EXTERNAL STORE DRAG INCREMENTS INCLUDED)

MACH	ALTITUDES - FEET						
	.0'	15000.0'	30000.0'	45000.0'	60000.0'	75000.0'	90000.0
.100	.01225	.01317	.01130	.01603	.01832	.02116	.02459
.300	.01019	.01091	.01180	.01313	.01488	.01702	.01956
.500	.00929	.00993	.01071	.01189	.01343	.01530	.01752
.700	.00859	.00918	.00990	.01098	.01239	.01409	.01609
.900	.00812	.00867	.00935	.01036	.01168	.01327	.01513
1.100	.02232	.02287	.02351	.02454	.02584	.02740	.02924
1.200	.02385	.02438	.02502	.02598	.02722	.02872	.03047
1.300	.02417	.02468	.02529	.02622	.02741	.02885	.03053
1.400	.02282	.02330	.02390	.02479	.02594	.02732	.02894
1.500	.02152	.02200	.02257	.02343	.02453	.02586	.02742
1.600	.02021	.02067	.02122	.02204	.02311	.02439	.02589
1.700	.01972	.02016	.02069	.02148	.02251	.02375	.02519
1.800	.01909	.01951	.02002	.02079	.02178	.02297	.02436
1.900	.01818	.01859	.01909	.01982	.02078	.02192	.02326
2.000	.01713	.01753	.01801	.01872	.01964	.02074	.02204

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

COEFFICIENT OF DRAG DUE-TO-LIFT

VALUES OF LIFT COEFFICIENT

MACH	.00000	.05000	.10000	.15000	.20000	.25000	.30000	.35000	.40000	.45000
.100	.00000	.00016	.00064	.00145	.00262	.00416	.00612	.00853	.01145	.01492
.300	.00000	.00016	.00066	.00150	.00270	.00428	.00628	.00873	.01169	.01520
.500	.00000	.00017	.00067	.00152	.00273	.00433	.00634	.00880	.01176	.01526
.700	.00000	.00017	.00069	.00156	.00280	.00444	.00649	.00900	.01200	.01554
.900	.00000	.00020	.00082	.00186	.00335	.00529	.00773	.01069	.01423	.01839
1.100	.00000	.00028	.00112	.00258	.00472	.00765	.01150	.01643	.02264	.03033
1.200	.00000	.00039	.00157	.00360	.00655	.01054	.01571	.02226	.03038	.04032
1.300	.00000	.00049	.00196	.00447	.00807	.01287	.01899	.02657	.03579	.04686
1.400	.00000	.00058	.00233	.00528	.00947	.01497	.02185	.03020	.04015	.05183
1.500	.00000	.00068	.00273	.00617	.01101	.01731	.02512	.03449	.04551	.05825
1.600	.00000	.00077	.00307	.00693	.01237	.01943	.02814	.03857	.05078	.06484
1.700	.00000	.00085	.00341	.00768	.01369	.02147	.03105	.04249	.05582	.07112
1.800	.00000	.00093	.00373	.00841	.01498	.02347	.03389	.04629	.06070	.07717
1.900	.00000	.00101	.00406	.00914	.01627	.02548	.03678	.05020	.06579	.08358
2.000	.00000	.00109	.00438	.00986	.01756	.02749	.03968	.05417	.07099	.09018

COEFFICIENT OF DRAG DUE-TO-LIFT

VALUES OF LIFT COEFFICIENT

MACH	.50000	.55000	.60000	.65000	.70000	.75000	.80000	.85000	.90000	.95000
.100	.01902	.02382	.02938	.03582	.04321	.05165	.06127	.07217	.08448	.09833
.300	.01932	.02113	.02969	.03608	.04341	.05175	.06122	.07192	.08397	.09749
.500	.01935	.02410	.02959	.03587	.04304	.05118	.06039	.07076	.08241	.09545
.700	.01967	.02446	.02996	.03624	.04340	.05149	.06063	.07090	.08240	.09524
.900	.02322	.02880	.03518	.04245	.05069	.05998	.07043	.08214	.09520	.10975
1.100	.03977	.05123	.06502	.08118	.10097	.12389	.15067	.18176	.21764	.25883
1.200	.05238	.06684	.08406	.10442	.12832	.15621	.18856	.22588	.26870	.31761
1.300	.06001	.07550	.09362	.11468	.13901	.16699	.19902	.23552	.27693	.32373
1.400	.06539	.08099	.09882	.11908	.14199	.16779	.19674	.22911	.26520	.30530
1.500	.07281	.08931	.10785	.12858	.15163	.17716	.20533	.23631	.27031	.30751
1.600	.08085	.09889	.11907	.14151	.16633	.19368	.22368	.25651	.29232	.33130
1.700	.08846	.10791	.12956	.15351	.17985	.20870	.24019	.27443	.31156	.35173
1.800	.09575	.11649	.13947	.16474	.19239	.22249	.25513	.29041	.32842	.36927
1.900	.10363	.12597	.15067	.17780	.20741	.23960	.27443	.31198	.35236	.39566
2.000	.11180	.13590	.16255	.19181	.22375	.25846	.29602	.33652	.38006	.42674

Figure 4-1. TAPE6 -- General Aircraft Output Data (Continued)

SUBSUB		GUESS = 1000.0000									
1 TAXI	2.00	0	.008	40000	.010	0	39160	.010	0	839
2 TAXI	.01	0	.083	39160	.010	0	38997	.010	0	163
3 ACCEL	2.00	1	.004	38997	.250	0	38440	.850	0	557
4 CLIMB	1.00	7	.016	38440	.850	0	38094	.850	25000	345
5 CRUISE	.43	491	.952	38094	.900	46492	35667	.900	47978	2427
6 LOITER	.26	0	1.000	35667	.690	36295	33508	.690	37594	2158
7 COMBAT	2.00	1	.004	33508	.800	10000	33056	.800	10000	451
9 CLIMB	1.00	3	.009	31056	.850	10000	30891	.850	25000	164
10 CRUISE	.42	496	.961	30891	.900	50393	28841	.900	52060	2050
11 LOITER	.06	0	.333	28841	.270	0	27828	.270	985	1012
4 CLIMB	1.00	28	.060	38440	.850	0	37736	.850	46492	703
5 CRUISE	.43	459	.891	37736	.900	46689	35478	.900	48088	2257
6 LOITER	.26	0	1.000	35478	.690	36295	33328	.690	37596	2150
7 COMBAT	2.00	1	.004	33328	.800	10000	32879	.800	10000	449
9 CLIMB	1.00	26	.055	30879	.850	10000	30377	.850	50393	502
10 CRUISE	.42	463	.899	30377	.900	50743	28483	.900	52320	1894
11 LOITER	.06	0	.333	28483	.270	0	27477	.270	991	1005
5 CRUISE	.43	449	.872	37736	.900	46689	35525	.900	48061	2210
6 LOITER	.26	0	1.000	35525	.690	36295	33373	.690	37595	2152
7 COMBAT	2.00	1	.004	33373	.800	10000	32923	.800	10000	449
9 CLIMB	1.00	26	.057	30923	.850	10000	30414	.850	50743	509
10 CRUISE	.42	453	.878	30414	.900	50718	28560	.900	52264	1854
11 LOITER	.06	0	.333	28560	.270	0	27553	.270	990	1007
5 CRUISE	.43	600	1.164	37736	.900	46689	34810	.900	48484	2926
6 LOITER	.25	0	1.000	34810	.690	36295	32690	.690	37603	2120
7 COMBAT	2.00	1	.004	32690	.800	10000	32249	.800	10000	440
9 CLIMB	1.00	25	.055	30249	.850	10000	29754	.850	50718	495
10 CRUISE	.42	604	1.172	29754	.900	51174	27345	.900	53350	2408
11 LOITER	.06	0	.333	27345	.270	0	26360	.270	1011	984
5 CRUISE	.43	595	1.154	37736	.900	46689	34833	.900	49471	2903
9 CLIMB	1.00	26	.057	30272	.850	10000	29770	.850	51174	501
10 CRUISE	.43	599	1.161	29770	.900	51163	27382	.900	53322	2388
11 LOITER	.06	0	.333	27382	.270	0	26396	.270	1010	985

MISSION : SUBSUB

											********	MID -	SEGMENT		PERFORMANCE		DATA	**	******
NO	SEGMENT	PS	D	I	WII	MI	ALTI	WIF	MF	ALIF	FUEL	WI	MACH	ALT	CL	L/D	FNAV	WFDOT	VKTAS	C.A.F.
1	TAXI	2.00	0	.008	40000	.010	0	39160	.010	0	839	39580	.010	0	.000	.00	46661	100746	6.6	6
2	TAXI	.01	0	.083	39160	.010	0	38997	.010	0	163	39078	.010	0	.000	.00	280	1964	6.6	6
3	ACCEL	2.00	1	.004	38997	.250	0	38440	.850	0	557	38726	.550	0	.151	14.18	52896	124140	363.8	445
4	CLIMB	1.00	28	.060	38440	.850	0	37736	.850	46492	703	38088	.850	18044	.112	11.28	18446	19585	526.2	457
5	CRUISE	.42	595	1.151	37736	.900	46689	34833	.900	48471	2903	36285	.900	47621	.412	16.00	5307	2516	516.2	310
6	LOITER	.25	0	1.000	34833	.690	36295	32712	.690	37603	2120	33750	.690	36945	.391	17.93	7394	2120	395.7	286
7	COMBAT	2.00	0	.004	32712	.800	10000	32272	.800	10000	440	32469	.800	10000	.997	7.95	46337	214385	510.6	448
8	DROP	2000	0	.000	32272	.800	10000	30272	.800	10000	0					STORE	DRAG	INDEX	IS NOW	0
9	CLIMB	1.00	26	.057	30272	.850	10000	29770	.850	51174	501	30021	.850	23044	.107	10.64	15407	15818	515.8	463
10	CRUISE	.41	599	1.161	29770	.900	51163	27382	.900	53322	2388	28576	.900	52434	.409	15.78	4261	2057	516.2	339
11	LOITER	.06	0	.333	27382	.270	0	26396	.270	1010	985	26889	.270	509	.444	17.60	27247	2956	178.3	170

RADIUS = 626 N.M.

Figure 4-1. TAPE6 — General Aircraft Output Data (Continued)

LEVEL FLIGHT PERFORMANCE

WEIGHT = 37680. POUNDS
80.0 PERCENT FUEL REMAINING

ALTITUDE = 0. FEET

MACH	CL	L/D	FNAVL	P.S.	SFC	RF	VDOT/G	PSUBS	LF	CAF
.600	.124	11.77	53730	.116	1.810	2579	1.341	53895	9.427	233
.700	.091	9.38	55811	.146	1.796	2417	1.375	64452	11.574	243
.800	.070	7.52	57892	.181	1.755	2267	1.404	75210	13.429	253
.900	.055	6.13	60274	.224	1.752	2083	1.437	86607	15.247	261
1.000	.045	2.73	62008	.585	1.609	1123	1.280	85731	14.167	333
1.100	.037	1.58	64814	1.020	1.452	791	1.087	80099	13.638	370
1.200	.031	1.25	67619	1.172	1.866	531	.994	79859	13.025	353

ALTITUDE = 20000. FEET

MACH	CL	L/D	FNAVL	P.S.	SFC	RF	VDOT/G	PSUBS	LF	CAF
.800	.151	13.26	34603	.167	1.549	4207	.843	41949	6.844	266
.900	.120	11.22	37271	.188	1.602	3872	.900	50394	7.918	276
1.000	.097	5.45	39703	.404	1.489	2249	.870	54138	7.535	345
1.100	.080	3.25	42036	.665	1.308	1679	.808	55300	7.510	383
1.200	.067	2.58	44369	.822	1.279	1487	.790	58993	7.515	395
1.300	.057	2.18	46356	.981	1.296	1346	.773	62474	7.708	400
1.400	.049	2.00	48341	1.048	1.428	1201	.782	68090	8.003	382
1.500	.043	1.84	50724	1.097	1.625	1044	.803	74943	8.196	364
1.600	.038	1.72	53103	1.135	1.784	948	.829	82486	8.417	345

ALTITUDE = 40000. FEET

MACH	CL	L/D	FNAVL	P.S.	SFC	RF	VDOT/G	PSUBS	LF	CAF
.800	.376	16.87	14878	.323	1.117	6928	.336	15593	2.820	309
.900	.297	16.37	16411	.305	1.212	6968	.374	19573	3.296	309
1.000	.241	10.06	18170	.482	1.265	4560	.383	22234	3.189	371
1.100	.199	6.70	20197	.612	1.176	3596	.387	24717	3.271	406
1.200	.167	5.41	22225	.715	1.171	3177	.405	28222	3.390	418
1.300	.142	4.65	23868	.798	1.207	2869	.418	31584	3.598	422
1.400	.123	4.28	25511	.834	1.237	2774	.443	36038	3.857	417
1.500	.107	3.97	26891	.882	1.266	2698	.462	40228	4.006	406
1.600	.094	3.74	28270	.921	1.301	2635	.483	44854	4.145	392
1.700	.083	3.43	29537	.985	1.319	2537	.493	48648	4.248	380

Figure 4-1. TAPE6 – General Aircraft Output Data (Continued)

** ENGINE DATA **

******** PROCESSING ENGINE NUMBER 621.

:: TI-FI JT69621 CONFIG.3:SUBSONIC/SUPERSONIC DASH CASE1 CARD2E

ALTITUDE-MACH NUMBER ARRAY

PS AUG .00 PS MIL 50.00 PS NR 50.00

ALTITUDES	0.	10000.	20000.	30000.	36089.	40000.	50000.	57500.	60000.	70000.
ALTITUDE	MACH	MACH	MACH	MACH	MACH	MACH	MACH	MACH	MACH	MACH
0.	.000	.200	.400	.600	.800	.900	1.000	1.200		
10000.	.400	.600	.800	.900	1.000	1.200				
20000.	.400	.600	.800	.900	1.000	1.200	1.400	1.600		
30000.	.400	.600	.800	.900	1.000	1.200	1.400	1.600	1.800	2.000
36089.	.600	.800	.900	1.000	1.200	1.400	1.600	1.800	2.000	
40000.	.600	.800	.900	1.000	1.200	1.400	1.600	1.800	2.000	
50000.	.600	.800	.900	1.200	1.600	2.000				
57500.	.600	.800	.900	1.200	1.600	2.000				
60000.	1.600	2.000								
70000.	1.600	2.000								

Figure 4-1. TAPE6 – General Aircraft Output Data (continued)

PROPULSION INLET TABLES
ATS2

*** TABLE 103 *** OPTIMUM INLET RECOVERY (PTO/PT OPT) VS LOCAL MACH NUMBER (MNO)

.000	.200	.400	.600	.800	1.000	1.200	1.600	2.000	MNO
.900	.950	.965	.972	.975	.975	.967	.948	.925	PT2/PTO

*** TABLE 104 *** OPTIMUM MASS FLOW RATIO (AO/AC OPT) VS LOCAL MACH NUMBER (MNO)

.400	.600	.800	1.000	1.200	1.600	2.000	MNO
1.310	.968	.863	.830	.840	.872	.915	AO/AC

*** TABLE 140 *** INLET DRAG COEFFICIENTS (CD) VS CORRECTED AIRFLOW / CAPTURE AREA (WC/AC) AND LOCAL MACH NUMBER (MNO)

MNO= .550	14.000	15.474	16.947	18.421	19.895	21.368	22.842	24.316	25.789	27.263	(WC/AC)
	28.737	30.211	31.684	33.158	34.632	36.105	37.579	39.053	40.526	42.000	
	.145	.117	.094	.072	.051	.037	.024	.013	.008	.004	(CD)
	.002	.001	.000	.000	.000	.000	.000	.000	.000	.000	
MNO= .600	14.000	15.474	16.947	18.421	19.895	21.368	22.842	24.316	25.789	27.263	(WC/AC)
	28.737	30.211	31.684	33.158	34.632	36.105	37.579	39.053	40.526	42.000	
	.198	.168	.139	.116	.092	.071	.055	.040	.028	.021	(CD)
	.015	.011	.008	.006	.004	.003	.002	.001	.001	.000	
MNO= .700	14.000	15.474	16.947	18.421	19.895	21.368	22.842	24.316	25.789	27.263	(WC/AC)
	28.737	30.211	31.684	33.158	34.632	36.105	37.579	39.053	40.526	42.000	
	.303	.270	.236	.204	.177	.149	.122	.103	.083	.063	(CD)
	.053	.043	.034	.028	.022	.017	.013	.010	.007	.005	
MNO= .800	14.000	15.474	16.947	18.421	19.895	21.368	22.842	24.316	25.789	27.263	(WC/AC)
	28.737	30.211	31.684	33.158	34.632	36.105	37.579	39.053	40.526	42.000	
	.387	.350	.313	.276	.242	.212	.182	.154	.132	.110	(CD)
	.088	.074	.061	.049	.040	.033	.027	.021	.018	.014	
MNO= .850	14.000	15.474	16.947	18.421	19.895	21.368	22.842	24.316	25.789	27.263	(WC/AC)
	28.737	30.211	31.684	33.158	34.632	36.105	37.579	39.053	40.526	42.000	
	.426	.387	.348	.309	.272	.240	.209	.177	.154	.131	(CD)
	.107	.090	.076	.061	.049	.042	.035	.028	.023	.019	
MNO= .900	14.000	15.474	16.947	18.421	19.895	21.368	22.842	24.316	25.789	27.263	(WC/AC)
	28.737	30.211	31.684	33.158	34.632	36.105	37.579	39.053	40.526	42.000	
	.442	.404	.365	.325	.288	.256	.224	.193	.168	.145	(CD)
	.121	.102	.087	.072	.057	.049	.041	.034	.027	.023	

Figure 4-1. TAPE6 — General Aircraft Output Data (continued)

MNO= 1.000	14.000	15.474	16.947	18.421	19.895	21.368	22.842	24.316	25.789	27.263	(WC/AC)
	28.737	30.211	31.684	33.158	34.632	36.105	37.579	39.053	40.526	42.000	**********
	.470	.431	.392	.352	.315	.282	.249	.217	.191	.167	(CD)
	.143	.121	.105	.089	.073	.062	.053	.043	.034	.029	**********
MNO= 1.200	14.000	15.474	16.947	18.421	19.895	21.368	22.842	24.316	25.789	27.263	(WC/AC)
	28.737	30.211	31.684	33.158	34.632	36.105	37.579	39.053	40.526	42.000	**********
	.512	.470	.428	.387	.349	.313	.277	.245	.218	.191	(CD)
	.164	.145	.126	.107	.090	.076	.062	.048	.041	.034	**********
MNO= 1.400	14.000	15.474	16.947	18.421	19.895	21.368	22.842	24.316	25.789	27.263	(WC/AC)
	28.737	30.211	31.684	33.158	34.632	36.105	37.579	39.053	40.526	42.000	**********
	.737	.693	.648	.603	.559	.514	.470	.425	.380	.336	(CD)
	.291	.247	.206	.170	.141	.113	.087	.063	.042	.037	**********
MNO= 1.600	14.000	15.474	16.947	18.421	19.895	21.368	22.842	24.316	25.789	27.263	(WC/AC)
	28.737	30.211	31.684	33.158	34.632	36.105	37.579	39.053	40.526	42.000	**********
	.727	.679	.630	.582	.533	.485	.436	.387	.339	.290	(CD)
	.242	.200	.166	.137	.111	.086	.070	.058	.058	.058	**********
MNO= 1.800	14.000	15.474	16.947	18.421	19.895	21.368	22.842	24.316	25.789	27.263	(WC/AC)
	28.737	30.211	31.684	33.158	34.632	36.105	37.579	39.053	40.526	42.000	**********
	.686	.630	.574	.518	.462	.406	.350	.294	.238	.192	(CD)
	.155	.125	.097	.075	.064	.059	.059	.059	.059	.059	**********
MNO= 2.000	14.000	15.474	16.947	18.421	19.895	21.368	22.842	24.316	25.789	27.263	(WC/AC)
	28.737	30.211	31.684	33.158	34.632	36.105	37.579	39.053	40.526	42.000	**********
	.614	.548	.482	.416	.350	.284	.218	.164	.123	.090	(CD)
	.061	.047	.042	.040	.040	.040	.040	.040	.040	.040	**********

Figure 4-1. TAPE6 – General Aircraft Output Data (continued)

TABLE 126 AFT-BODY DRAG COEFFICIENT (CD A/B) VS NOZZLE PRESSURE RATIO (PS9/PAMB) AND NOZZLE AREA RATIO (A9/A8) AND AFT-BODY AREA RATIO (A9/A10)

PS9/PA = .500

A9/A											
A9/A= .549	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.039	.041	.048	.110	.098	.089	.066	.056	.052	.050	CD A/B
A9/A= .500	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.039	.040	.048	.116	.101	.091	.067	.058	.054	.052	CD A/B
A9/A= .400	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.039	.040	.045	.133	.119	.108	.088	.078	.074	.072	CD A/B
A9/A= .300	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.038	.039	.048	.159	.139	.127	.111	.102	.098	.095	CD A/B
A9/A= .200	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.041	.042	.053	.198	.175	.163	.149	.141	.136	.132	CD A/B
A9/A= .100	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.043	.044	.054	.249	.229	.215	.205	.198	.194	.190	CD A/B

PS9/PA = 1.000

A9/A											
A9/A= .549	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.009	.009	.014	.076	.066	.059	.056	.054	.052	.050	CD A/B
A9/A= .500	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.010	.010	.015	.083	.070	.063	.0[illegible]8	.056	.054	.052	CD A/B
A9/A= .400	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.015	.015	.018	.106	.093	.084	.080	.076	.074	.072	CD A/B
A9/A= .300	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.017	.017	.024	.135	.116	.106	.104	.100	.098	.095	CD A/B
A9/A= .200	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS

Figure 4-1. TAPE6 – General Aircraft Output Data (continued)

	.025	.025	.035	.180	.158	.147	.144	.140	.136	.132	CD A/B
A9/A= .100	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.035	.035	.045	.240	.220	.207	.202	.197	.194	.190	CD A/B
		PS9/PA = 1.500									
		***************									
A9/A= .549	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	-.021	-.023	-.020	.042	.034	.029	.046	.052	.052	.050	CD A/B
A9/A= .500	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	-.019	-.020	-.018	.050	.039	.035	.049	.054	.054	.052	CD A/B
A9/A= .400	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	-.009	-.010	-.009	.079	.067	.060	.072	.074	.074	.072	CD A/B
A9/A= .300	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	-.004	-.005	.000	.111	.093	.085	.097	.098	.098	.095	CD A/B
A9/A= .200	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.009	.008	.017	.162	.141	.131	.139	.139	.136	.132	CD A/B
A9/A= .100	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.027	.026	.036	.231	.211	.139	.199	.196	.194	.190	CD A/B
		PS9/PA = 2.000									
		***************									
A9/A= .549	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	-.052	-.054	-.054	.008	.001	.000	.037	.049	.051	.050	CD A/B
A9/A= .500	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	-.049	-.051	-.050	.018	.008	.006	.039	.052	.053	.052	CD A/B
A9/A= .400	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	-.034	-.035	-.036	.052	.041	.037	.065	.072	.073	.072	CD A/B
A9/A= .300	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	-.026	-.027	-.024	.087	.071	.064	.090	.097	.097	.095	CD A/B
A9/A= .200	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	-.008	-.009	-.001	.143	.123	.115	.134	.138	.136	.132	CD A/B
A9/A= .100	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.018	.018	.026	.221	.202	.191	.197	.196	.194	.190	CD A/B

Figure 4-1. TAPE6 – General Aircraft Output Data (continued)

PS9/PA = 3.000

A9/A= .549	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	-.113	-.117	-.121	-.060	-.064	-.059	.017	.045	.050	.050	CD A/B
A9/A= .500	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	-.107	-.111	-.115	-.048	-.055	-.051	.021	.047	.052	.052	CD A/B
A9/A= .400	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	-.082	-.086	-.090	-.003	-.011	-.011	.049	.069	.073	.072	CD A/B
A9/A= .300	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	-.068	-.072	-.071	.040	.025	.023	.077	.094	.097	.095	CD A/B
A9/A= .200	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	-.040	-.043	-.038	.107	.088	.083	.123	.135	.135	.132	CD A/B
A9/A= .100	.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNFS
	.001	.000	.008	.202	.184	.174	.191	.194	.194	.190	CD A/B

* TABLE 129 * LOCAL MACH NUMBER (MNO) VS DEL. AFT-BODY DRAG COEF. (DEL. A/B CD)

.600	.800	.900	1.000	1.100	1.200	1.600	2.000	2.400	3.000	MNO
.020	.020	.028	.153	.133	.123	.120	.116	.114	.110	CD A/B

Figure 4-1. TAPE6 – General Aircraft Output Data (continued)

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                         PROPULSION CFG TABLES
                                CV2DCD

************
* TABLE130* GROSS THRUST COEFFICIENT (CFG)          VS NOZZLE PRESSURE RATIO (PT9/PAMB)          AND NOZZLE AREA RATIO (A9/A8)
************

A9/A8 = 2.000 2.000    3.000    4.000    5.000    6.000    6.500    PT9/PAMB
               .989     .989     .987     .984     .977     .971    CFG

A9/A8 = 1.000 2.000    3.000    4.000    5.000    6.000    6.500    PT9/PAMB
               .985     .985     .987     .990     .993     .993    CFG


OPERATING REFERANCE
    .0000     .3000    .6000    .6010    .8000    .9000   1.0000   1.2000   1.4000   1.6000
   1.8000    2.0000
    .0000     .0000 .000036089.000036089.000036089.000036089.000036089.000036089.000036089.0000
6089.000036089.0000
   2.0000    2.0000   2.0000   2.0000   2.0000   2.0000   2.0000   2.0000   2.0000   2.0000
   2.0000    2.0000
```

Figure 4-1. TAPE6 — General Aircraft Output Data (continued)

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CAPTURE AREA  5.67          SIZE POINT  MACH  2.00     ALTITUDE36089.
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Figure 4-1. TAPE6 — General Aircraft Output Data (continued)

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INSTALLED PROPULSION SUB-SYSTEM PERFORMANCE OUTPUT MAPS FOR SINGLE ENGINE                    DATE RUN  86/04/04.
                                  ::::::::  TI-FI JT69621 CONFIG.3:SUBSONIC/SUPERSONIC DASH CASE1        CARD2E
LIBRARY NO.           0  YEAR              1989.  TOGW                0.  SCALE           .9295  WT ENGINE        0.
FN SLS MIL      15065.  FN SLS MAX        25001.  BYPASS RATIO      1.00  T4 DESIGN       3000.  T7 DESIGN        0.  PRESSURE RATIO 25.00
ACAPT           5.270  ACC               11.473  FRT FLANGE DIA     .00  INLET LENGTH      .00  DUCT LENGTH     .00  EXH LENGTH       .00
AIRPLANE INTERFACE%  NO. OF ENGINES       2.  A10              41.112  ACCT/A10         .558  T/W UNINST MAX  .000  T/W INST MAX    .000
POWER SETTING CODE% (PS=O. TO 1.0  PCT OF MIL PWR)  (PS=1.0 TO 2.0  PCT OF AUG REG)

ALTITUDE      O.      MACH NO.   .000
OPERATING REFERENCE CONDITION          ALTITUDE      O.  MACH NO.      .000  POWER SETTING 2.00  A9/A10    .228  PS9/PO    .964
                                                                                               CDINL    .0684  CDAFT    .0238
```

1. UNINSTALLED (INPUT) ENGINE PERFORMANCE

FNU	WF	WO COR	SFC	FGI	CFG	RF
25000.	49356.	215.9	1.974	25407.	.984	1.000
15536.	12251.	215.9	.789	15788.	.984	1.000
15065.	10894.	215.9	.723	15309.	.984	1.000
14255.	10061.	211.0	.706	14487.	.984	1.000
9919.	6496.	177.3	.655	10070.	.985	1.000
5452.	3581.	138.4	.657	5697.	.957	1.000
2398.	1827.	95.6	.762	2445.	.981	1.000

2. INLET SYSTEM PERFORMANCE

RF	AOE/AC	AO/AC	WCAC	CDPS	DPS/FN	DREF/FN
.900	.000	.000	47.617	.2133	.0000	.0000
.900	.000	.000	47.617	.2133	.0000	.0000
.900	.000	.000	47.617	.2133	.0000	.0000
.900	.000	.000	46.531	.2133	.0000	.0000
.900	.000	.000	39.110	.2133	.0000	.0000
.900	.000	.000	30.522	.2133	.0000	.0000
.900	.000	.000	21.072	.2133	.0000	.0000

3. AFTERBODY/EXHAUST SYSTEM PERFORMANCE   **FN INCLUDES POWER SENSITIVE DRAG ONLY*

PT8/PO	PS9/PO	A8	A9	CFGIN	A9/ACC	A9T/A10	CDPS	DPS/FN	DREF/FN
2.763	.964	4.318	4.677	.989	.439	.228	.0000	.0000	.0000
2.880	1.027	2.598	2.652	.985	.249	.129	.0079	.0000	.0000
2.889	1.007	2.512	2.565	.985	.240	.125	.0086	.0000	.0000
2.772	.970	2.512	2.563	.985	.240	.125	.0091	.0000	.0000
2.151	.920	2.512	2.432	.984	.228	.118	.0104	.0000	.0000
1.521	1.119	2.586	2.639	.982	.247	.128	.0066	.0000	.0000
1.125	1.043	3.183	3.846	.982	.361	.187	.0014	.0000	.0000

4. INSTALLED PROPULSION SYSTEM PERFORMANCE

PS	FN	SFC	FRAM	FNC	SFCC
2.00	21640.	2.053	0.	21640.	2.053
1.05	13437.	.821	0.	13437.	.821
1.00	13030.	.752	0.	13030.	.752
.94	12305.	.736	0.	12305.	.736
.65	8415.	.695	0.	8415.	.695
.35	4535.	.711	0.	4535.	.711
.12	1581.	1.040	0.	1581.	1.041

**FN INCLUDES POWER SENSITIVE DRAG ONLY*

5. SECOND CARD - UNINSTALLED SPECIAL PARAMETERS

T4	WC2	P8QAMB	NI	TC	TH	TP	BPR	OPR	BPR
3460.	232.	3.067	11633.	803	2097.	3741	3.552	27.214	.966
3460.	232.	3.201	11633.	803	2097.	1601	3.552	27.214	.966
3460.	232	3.206	11633.	803	2097	1506	3.552	27.214	.966
3360	227.	3.078	11413	789	2035.	1458	3.421	25.924	.986
2936.	191	2.386	10335.	729	1778	1258	2.703	19.129	1.098
2418.	149.	1.688	8769.	670	1486.	1051	1.965	12.677	1.235
2008.	103.	1.250	6609.	620	1260.	915	1.415	7.868	1.233

Figure 4-1. TAPE6 – General Aircraft Output Data (continued)

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     INSTALLED PROPULSION SUB-SYSTEM PERFORMANCE OUTPUT MAPS FOR SINGLE ENGINE                            DATE RUN  86/04/04.
                                            :::::::: TI-FI JT69621 CONFIG.3:SUBSONIC/SUPERSONIC DASH CASE1        CARD2E
LIBRARY NO.         0  YEAR              1989.  TOGW             0.  SCALE             .9295  WT ENGINE         0.
FN SLS MIL    15065.  FN SLS MAX       25001.  BYPASS RATIO    1.00  T4 DESIGN        3000.  T7 DESIGN         0.  PRESSURE RATIO 25.00
ACAPT          5.270  ACC             11.473  FRT FLANGE DIA   .00  INLET LENGTH       .00  DUCT LENGTH      .00  EXH LENGTH        .00
AIRPLANE INTERFACE%  NO. OF ENGINES     2.  A10           41.112  ACCT/A10          .558  T/W UNINST MAX  .000  T/W INST MAX     .000
POWER SETTING CODE% (PS=0. TO 1.0  PCT OF MIL PWR)  (PS=1.0 TO 2.0  PCT OF AUG REG)

ALTITUDE70000.      MACH NO. 2.000
OPERATING REFERENCE CONDITION          ALTITUDE    36089.  MACH NO.     2.000  POWER SETTING 2.00  A9/A10    .419  PS9/PO   1.478
                                                                                                  CDINL    .0684  CDAFT    .0702

1. UNINSTALLED (INPUT) ENGINE PERFORMANCE                          2. INLET SYSTEM PERFORMANCE
    FNU     WF   WO COR    SFC     FGI      CFG     RF                RF   AOE/AC AO/AC  WCAC     CDPS   DPS/FN DREF/FN
  3329.  7985.   131.3  2.398   5669.    .966   .925              .925    .915   .915 28.964 -.0095  -.0037   .0268
  2724.  5402.   131.3  1.983   5027.    .969   .925              .925    .915   .915 28.964 -.0095  -.0045   .0324
  1881.  3144.   131.3  1.671   4153.    .970   .925              .925    .915   .915 28.964 -.0095  -.0064   .0461
  1222.  1705.   131.3  1.395   3455.    .975   .925              .925    .915   .915 28.964 -.0095  -.0107   .0767
  1101.  1483.   131.3  1.348   3351.    .969   .925              .925    .915   .915 28.964 -.0095  -.0112   .0800
   993.  1331.   125.2  1.341   3117.    .975   .925              .925    .872   .915 27.611  .0148   .0216   .1001
   685.   976.   109.2  1.425   2528.    .977   .925              .925    .761   .915 24.085  .1041   .3815   .2507
   444.   719.    97.1  1.617   2088.    .973   .925              .925    .677   .915 21.420  .2133 -4.6870 -1.5029

3. AFTERBODY/EXHAUST SYSTEM PERFORMANCE    **FN INCLUDES POWER SENSITIVE DRAG ONLY*     4. INSTALLED PROPULSION SYSTEM PERFORMANCE
  PT8/PO PS9/PO    A8      A9    CFGIN A9/ACC A9T/A10  CDPS   DPS/FN DREF/FN                PS     FN     SFC    FRAM    FNC    SFCC
 11.430  1.479  4.294   7.884   .912   .739   .384  .0078   .0139   .1248              2.00  2992.  2.669  2147. 68529.  3.068
 11.600  1.219  3.751   7.455   .932   .699   .363  .0139   .0298   .1507              1.10  2478.  2.180  2147. 56748.  2.507
 11.790  1.006  3.080   6.712   .963   .629   .327  .0234   .0715   .2147              1.05  1739.  1.807  2147. 39834.  2.078
 11.910  1.009  2.591   5.437   .988   .510   .264  .0440   .2236   .3572              1.00  1045.  1.631  2147. 23943.  1.875
 11.930   .817  2.511   5.901   .993   .553   .287  .0356   .1891   .3725              1.00  1003.  1.480  2147. 22961.  1.701
 11.180   .927  2.512   5.276  1.000   .495   .257  .0474   .3142   .4659               .80   801.  1.661  2046. 18356.  1.909
  9.320   .908  2.512   4.801  1.000   .450   .234  .0566   .9408  1.1669               .32   320.  3.050  1785.  7329.  3.506
  7.910   .772  2.512   4.793  1.000   .449   .233  .0571 -5.6920 -6.9953              -.05   -53.-13.459  1588.  -1223.-15.474

5. SECOND CARD - UNINSTALLED SPECIAL PARAMETERS                                       **FN INCLUDES POWER SENSITIVE DRAG ONLY*
 T4 WC2 P8OAMB N1 TC  TH TP BPR OPR BPR
  3460.     141.  11.427  9995.     906.  2184.   3665.   1.909 11.895   1.272
  3160.     141.  11.601  9995.     906.  2184.   3008.   1.909 11.895   1.272
  3460.     141.  11.794  9995.     906.  2184.   2159.   1.909 11.895   1.272
  3460.     141.  11.911  9995.     906.  2184.   1591.   1.909 11.895   1.272
  3460.     141.  11.928  9995.     906.  2184.   1503.   1.909 11.895   1.272
  3360.     135.  11.178  9642.     895.  2123.   1461.   1.799 11.008   1.295
  3092.     118.   9.323  8648.     864.  1962.   1348.   1.522  8.808   1.375
  2838.     105.   7.913  7805.     832.  1812.   1241.   1.316  7.087   1.485
```

Figure 4-1. TAPE6 – General Aircraft Output Data (continued)

AIRPLANE INLET OUTPUT MAPS

```
* POWER SETTING CODE *
TI-FI JT6962I CONFIG.3:SUBSONI
*                      *
*  PS = 0.0 TO 1.0     *     TOGW            0.       ACAPT        5.270    A10                41.112
*PERCENT OF MIL POWER*
*                      *     ENGINE SCALE   .9295     MACH SIZE    2.000    AWET(FOREBODY)       .00
*  PS = 1.0 TO 2.0     *
*PERCENT OF AUG REG   *                               ALT SIZE     36089.   AWET(AFTBODY)        .00
```

1. MAXIMUM INLET SUPPLY AND MAXIMUM ENGINE DEMAND FLOW SCHEDULES FOR INLET SIZING (FOR SINGLE ENGINE)

MACH NUMBER	.800	.900	1.000	1.200	1.400	1.600	1.800	2.000
*ENGINE MAXIMUM AIRFLOW								
WOC MAX	217.50	217.50	217.50	217.50	207.09	189.43	169.45	152.62
ALTITUDE	57500.	57500.	40000.	36089.	36089.	36089.	36089.	36089.
*INLET MAXIMUM FLOW SCHEDULES								
AO/AC (MAX REC)	.863	.847	.830	.840	.856	.872	.894	.915
RF (MAX REC)	.975	.975	.975	.967	.958	.948	.937	.925
*INLET/ENGINE MATCH AT MAX FLOW								
AO ENG REQ	4.456	4.330	4.292	4.386	4.474	4.544	4.621	4.822
AC REQ	5.163	5.115	5.171	5.222	5.227	5.211	5.172	5.270
*ENGINE MAX DEMAND MASS FLOW RATIO SCHEDULE								
RF	.975	.975	.975	.967	.958	.948	.937	.925
AOE/ACAPT	.846	.822	.814	.832	.849	.862	.877	.915
AOBYP/ACAPT	.017	.025	.016	.008	.007	.010	.016	.000

2. AIRPLANE INLET OPERATING REFERENCE CONDITIONS
- RUBBER INLET -
(FOR SINGLE ENGINE)

MACH NUMBER	.000	.200	.300	.400	.600	.601	.800	.900	1.000	1.200	1.400	1.600	1.800	2.000
POWER SETTING	2.000*******		2.000*******		2.000	2.000	2.000	2.000	2.000	2.000	2.000	2.000	2.000	2.000
ALTITUDE	0.*******		0.*******		0.	36089.	36089.	36089.	36089.	36089.	36089.	36089.	36089.	36089.
WO CORR.	215.92	214.00	213.04	210.47	205.33	217.50	217.50	217.50	217.50	217.50	207.09	189.43	169.45	152.62
AOE/ACAPT	.000	.000	.000	.304	.911	.964	.863	.847	.830	.840	.856	.872	.894	.915
RF	.900	.938	.958	.962	.972	.972	.975	.975	.975	.967	.958	.948	.937	.925
CD(INL OPER REF)	.0000	.0000	.0000	.0004	.0011	.0005	.0137	.0216	.0288	.0356	.0549	.0833	.0809	.0588

Figure 4-1. TAPE6 – General Aircraft Output Data (continued)

AIRPLANE INLET OUTPUT MAPS (CONTINUED)

3. INLET DRAG INCLUDED IN THE AIRFRAME SUB-SYSTEM FORCE (POLAR) - RUBBER INLET -

CD (INL) = D(INL) / (Q * ACAPT)

MACH NUMBER	.000	.300	.600	.601	.800	.900	1.000	1.200	1.400	1.600	1.800	2.000
*MFR=1 TO CRIT DELTA CD(DIV,COWL)	.0000	.0000	.0000	.0000	.0000	.0000	.0000	.0000	.0000	.0000	.0000	.0000
CD(INL CRIT)	.0000	.0000	.0000	.0000	.0000	.0000	.0000	.0000	.0000	.0000	.0000	.0000
*MFR=CRIT TO OPER REF CD(INL OPER REF)	.0000	.0000	.0011	.0005	.0137	.0216	.0288	.0356	.0549	.0833	.0809	.0588
AOE/ACAPT (OPER REF)	.000	.000	.911	.964	.863	.847	.830	.840	.856	.872	.894	.915

Figure 4-1. TAPE6 – General Aircraft Output Data (continued)

AIRPLANE INLET OUTPUT MAPS (CONTINUED)

7. INLET/ENGINE MATCH AT MAX FLOW - RUBBER INLET -

MACH \ ALT	0.	10000.	20000.	30000.	36089.	40000.	50000.	57500.	60000.	70000.
.000	5.270	*******	*******	*******	*******	*******	*******	*******	*******	*******
.200	5.270	*******	*******	*******	*******	*******	*******	*******	*******	*******
.300	*******	*******	*******	*******	*******	*******	*******	*******	*******	*******
.400	5.270	5.270	5.270	5.270	*******	*******	*******	*******	*******	*******
.600	5.270	5.270	5.270	5.270	5.270	5.270	5.270	5.270	*******	*******
.601	*******	*******	*******	*******	*******	*******	*******	*******	*******	*******
.800	4.612	4.945	5.163	5.163	5.163	5.163	5.163	5.163	*******	*******
.900	4.409	4.768	5.017	5.115	5.115	5.115	5.115	5.115	*******	*******
1.000	4.274	4.654	4.996	5.171	5.171	5.171	*******	*******	*******	*******
1.200	3.948	4.300	4.713	5.074	5.222	5.215	5.099	4.972	*******	*******
1.400	*******	*******	4.493	4.962	5.227	5.197	*******	*******	*******	*******
1.600	*******	*******	4.418	4.871	5.211	5.165	4.989	4.792	4.712	4.250
1.800	*******	*******	*******	4.863	5.172	5.133	*******	*******	*******	*******
2.000	*******	*******	*******	4.971	5.270	5.234	5.093	4.949	4.891	4.535

Figure 4-1. TAPE6 – General Aircraft Output Data (continued)

AIRPLANE INLET OUTPUT MAPS (CONTINUED)

8. ENGINE MAX AIR FLOW - RUBBER INLET -

MACH \ ALT	0.	10000.	20000.	30000.	36089.	40000.	50000.	57500.	60000.	70000.
.000	215.9	*******	*******	*******	*******	*******	*******	*******	*******	*******
.200	214.6	*******	*******	*******	*******	*******	*******	*******	*******	*******
.300	*******	*******	*******	*******	*******	*******	*******	*******	*******	*******
.400	211.5	217.5	217.5	217.5	*******	*******	*******	*******	*******	*******
.600	205.3	215.8	217.5	217.5	217.5	217.5	217.5	217.5	*******	*******
.601	*******	*******	*******	*******	*******	*******	*******	*******	*******	*******
.800	194.3	208.3	217.5	217.5	217.5	217.5	217.5	217.5	*******	*******
.900	187.5	202.7	214.6	217.5	217.5	217.5	217.5	217.5	*******	*******
1.000	179.8	195.8	210.2	217.5	217.5	217.5	*******	*******	*******	*******
1.200	164.4	179.1	196.3	211.4	217.5	217.2	212.4	207.1	*******	*******
1.400	*******	*******	178.0	196.6	207.1	205.9	*******	*******	*******	*******
1.600	*******	*******	160.6	177.1	189.4	187.8	181.3	174.2	171.3	154.5
1.800	*******	*******	*******	159.3	169.4	168.1	*******	*******	*******	*******
2.000	*******	*******	*******	144.0	152.6	151.6	147.5	143.3	141.7	131.3

Figure 4-1. TAPE6 — General Aircraft Output Data (continued)

AIRPLANE AFTERBODY OUTPUT MAPS

```
* POWER SETTING CODE *
TI-FI JT69621 CONFIG.3:SUBSONI
*                    *
*  PS = 0.0 TO 1.0   *      TOGW              0.      A10   41.112    ACC/A10   .5581    AWET(FOREBODY)    .00
*PERCENT OF MIL POWER*
*                    *      ENGINE SCALE   .9295      ACC   22.946    LCC/D10   .0000    AWET(AFTBODY)     .00
*  PS = 1.0 TO 2.0   *
*PERCENT OF AUG REG  *                          CD(AFT) = D(AFT) /( Q * A10 )
```

1. AIRPLANE AFTERBODY OPERATING REFERENCE CONDITIONS

MACH NUMBER	.000	.200	.300	.400	.600	.601	.800	.900	1.000	1.200	1.400	1.600	1.800	2.000
POWER SETTING	2.000	*******	2.000	*******	2.000	2.000	2.000	2.000	2.000	2.000	2.000	2.000	2.000	2.000
ALTITUDE	0.	*******	0.	*******	0.	36089.	36089.	36089.	36089.	36089.	36089.	36089.	36089.	36089.
A9/A10	.2275	.2288	.2295	.2350	.2459	.2857	.3088	.3059	.3304	.3412	.3620	.3878	.3872	.4194
A9/ACC	.4077	.4100	.4111	.4210	.4407	.5119	.5533	.5481	.5920	.6113	.6486	.6949	.6938	.7514
PS9/PO	.9644	1.0253	1.0558	1.0424	1.0157	.9915	.9644	1.0590	.9962	1.1227	1.2092	1.2555	1.4482	1.4777
CD(AFT OPER REF)	.0238	.0219	.0209	.0208	.0207	.0185	.0184	.0208	.1264	.0915	.0844	.0789	.0766	.0702

2. AFTERBODY POWER SENSITIVE DRAG MAP

DELTA CD(AFT PS) = CD(AFT OPER) - CD(AFT OPER REF)

PS9/PO= .5000

OPERATING RANGE A9/A10	A9/ACC	MACH .000	.300	.600	.601	.800	.900	1.000	1.200	1.400	1.600	1.800	2.000
.1000	.1792	.0162	.0206	.0223	.0245	.0256	.0332	.1226	.1235	.1256	.1261	.1249	.1278
.2000	.3583	.0142	.0186	.0203	.0225	.0236	.0322	.0716	.0715	.0716	.0701	.0684	.0708
.3000	.5375	.0112	.0156	.0173	.0195	.0206	.0272	.0326	.0355	.0346	.0321	.0299	.0318
.4000	.7167	.0122	.0166	.0183	.0205	.0216	.0242	.0066	.0165	.0136	.0091	.0064	.0078
.5000	.8959	.0122	.0166	.0183	.0205	.0216	.0272	-.0104	-.0005	-.0054	-.0119	-.0141	-.0122
.6000	1.0750	.0061	.0136	.0183	.0205	.0236	.0272	-.0226	-.0046	-.0084	-.0139	-.0172	-.0163
.7000	1.2542	-.0001	.0105	.0183	.0205	.0257	.0272	-.0349	-.0086	-.0115	-.0160	-.0203	-.0204
.8000	1.4334	-.0062	.0075	.0183	.0205	.0277	.0272	-.0471	-.0127	-.0145	-.0180	-.0233	-.0245
.9000	1.6125	-.0123	.0044	.0183	.0206	.0298	.0272	-.0593	-.0168	-.0176	-.0201	-.0264	-.0285

PS9/PO= 1.0000

OPERATING RANGE A9/A10	A9/ACC	MACH .000	.300	.600	.601	.800	.900	1.000	1.200	1.400	1.600	1.800	2.000
.1000	.1792	.0112	.0141	.0143	.0165	.0166	.0242	.1136	.1155	.1201	.1231	.1229	.1268
.2000	.3583	.0012	.0041	.0043	.0065	.0066	.0142	.0536	.0555	.0611	.0651	.0654	.0698
.3000	.5375	-.0068	-.0039	-.0037	-.0015	-.0014	.0032	.0086	.0145	.0206	.0251	.0254	.0298
.4000	.7167	-.0088	-.0059	-.0057	-.0035	-.0034	-.0028	-.0204	-.0075	-.0024	.0011	.0014	.0058
.5000	.8959	-.0138	-.0109	-.0107	-.0085	-.0084	-.0058	-.0434	-.0285	-.0239	-.0209	-.0196	-.0142
.6000	1.0750	-.0159	-.0129	-.0128	-.0105	-.0104	-.0078	-.0577	-.0366	-.0300	-.0250	-.0237	-.0183
.7000	1.2542	-.0179	-.0149	-.0148	-.0126	-.0125	-.0099	-.0719	-.0448	-.0361	-.0291	-.0278	-.0224
.8000	1.4334	-.0199	-.0170	-.0169	-.0146	-.0145	-.0119	-.0862	-.0530	-.0422	-.0331	-.0319	-.0265
.9000	1.6125	-.0220	-.0190	-.0189	-.0167	-.0166	-.0140	-.1005	-.0611	-.0484	-.0372	-.0360	-.0305

Figure 4-1. TAPE6 — General Aircraft Output Data (continued)

AIRPLANE AFTERBODY OUTPUT MAPS (CONTINUED)

PS9/PO= 1.5000

OPERATING RANGE A9/A10	A9/ACC	MACH .000	.300	.600	.601	.800	.900	1.000	1.200	1.400	1.600	1.800	2.000
.1000	.1792	.0062	.0076	.0063	.0085	.0076	.0152	.1046	.1075	.1146	.1201	.1209	.1258
.2000	.3583	-.0118	-.0104	-.0117	-.0095	-.0104	-.0038	.0356	.0395	.0506	.0601	.0624	.0688
.3000	.5375	-.0248	-.0234	-.0247	-.0225	-.0234	-.0208	-.0154	-.0065	.0066	.0181	.0209	.0278
.4000	.7167	-.0298	-.0284	-.0297	-.0275	-.0284	-.0298	-.0474	-.0315	-.0184	-.0069	-.0036	.0038
.5000	.8959	-.0398	-.0384	-.0397	-.0375	-.0384	-.0388	-.0764	-.0565	-.0424	-.0299	-.0251	-.0162
.6000	1.0750	-.0378	-.0394	-.0438	-.0416	-.0445	-.0429	-.0927	-.0687	-.0515	-.0360	-.0302	-.0203
.7000	1.2542	-.0357	-.0404	-.0479	-.0457	-.0507	-.0470	-.1090	-.0810	-.0607	-.0421	-.0353	-.0244
.8000	1.4334	-.0337	-.0414	-.0520	-.0498	-.0568	-.0510	-.1253	-.0932	-.0699	-.0483	-.0405	-.0285
.9000	1.6125	-.0317	-.0424	-.0561	-.0539	-.0629	-.0551	-.1417	-.1055	-.0791	-.0544	-.0456	-.0325

PS9/PO= 2.0000

OPERATING RANGE A9/A10	A9/ACC	MACH .000	.300	.600	.601	.800	.900	1.000	1.200	1.400	1.600	1.800	2.000
.1000	.1792	-.0058	-.0029	-.0027	-.0005	-.0004	.0052	.0946	.0995	.1096	.1181	.1199	.1258
.2000	.3583	-.0288	-.0274	-.0287	-.0265	-.0274	-.0218	.0166	.0235	.0401	.0551	.0594	.0678
.3000	.5375	-.0468	-.0454	-.0467	-.0445	-.0454	-.0448	-.0394	-.0275	-.0074	.0111	.0169	.0268
.4000	.7167	-.0548	-.0534	-.0547	-.0525	-.0534	-.0568	-.0744	-.0545	-.0334	-.0139	-.0081	.0018
.5000	.8959	-.0668	-.0669	-.0697	-.0675	-.0694	-.0708	-.1084	-.0855	-.0619	-.0399	-.0311	-.0182
.6000	1.0750	-.0729	-.0730	-.0759	-.0736	-.0755	-.0790	-.1288	-.0977	-.0700	-.0440	-.0362	-.0243
.7000	1.2542	-.0791	-.0791	-.0820	-.0797	-.0817	-.0871	-.1492	-.1100	-.0782	-.0481	-.0413	-.0305
.8000	1.4334	-.0852	-.0852	-.0881	-.0859	-.0878	-.0953	-.1696	-.1222	-.0864	-.0521	-.0465	-.0366
.9000	1.6125	-.0913	-.0914	-.0942	-.0920	-.0939	-.1034	-.1900	-.1345	-.0945	-.0562	-.0516	-.0427

PS9/PO= 3.0000

OPERATING RANGE A9/A10	A9/ACC	MACH .000	.300	.600	.601	.800	.900	1.000	1.200	1.400	1.600	1.800	2.000
.1000	.1792	-.0198	-.0184	-.0197	-.0175	-.0184	-.0128	.0756	.0825	.0981	.1121	.1159	.1238
.2000	.3583	-.0548	-.0564	-.0607	-.0585	-.0614	-.0588	-.0194	-.0085	.0186	.0441	.0524	.0648
.3000	.5375	-.0798	-.0829	-.0887	-.0865	-.0904	-.0918	-.0864	-.0685	-.0344	-.0019	.0089	.0238
.4000	.7167	-.0938	-.0969	-.1027	-.1005	-.1044	-.1108	-.1294	-.1025	-.0654	-.0299	-.0176	-.0012
.5000	.8959	-.1188	-.1219	-.1277	-.1255	-.1294	-.1358	-.1744	-.1425	-.0994	-.0579	-.0426	-.0232
.6000	1.0750	-.1311	-.1341	-.1400	-.1378	-.1417	-.1480	-.1989	-.1588	-.1116	-.0661	-.0488	-.0273
.7000	1.2542	-.1433	-.1464	-.1522	-.1500	-.1539	-.1603	-.2233	-.1751	-.1239	-.0742	-.0549	-.0314
.8000	1.4334	-.1556	-.1586	-.1645	-.1622	-.1661	-.1725	-.2478	-.1915	-.1361	-.0824	-.0610	-.0355
.9000	1.6125	-.1678	-.1708	-.1767	-.1745	-.1784	-.1848	-.2723	-.2078	-.1483	-.0905	-.0671	-.0395

Figure 4-1. TAPE6 — General Aircraft Output Data (continued)

AIRPLANE AFTERBODY OUTPUT MAPS (CONTINUED)

3. AFTERBODY REFERENCE DRAG COEFFICIENTS

MACH NUMBER	.000	.300	.600	601	800	.900	1 000	1.200	1.400	1.600	1.800	2.000
CD(AFT OPER REF)	.0238	.0209	.0207	.0185	0184	.0208	.1264	.0915	.0844	.0789	.0766	.0702
CD(AFT AERO REF)	.0203	.0203	.0203	.0203	0203	.0285	.1534	.1228	1216	.1204	.1184	.1164
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DELTA CD(AFT REF)	.0035	.0006	.0005	-.0018	-.0019	-.0077	-.0270	-.0313	-.0372	-.0415	-.0417	-.0461

Figure 4-1. TAPE6 — General Aircraft Output Data (continued)

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